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Atmospheric, Magnetospheric, and Plasmas in Space (AMPS) Spacelab Payload Definition Study

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ATMOSPHERIC, MAGNETOSPHERIC AND PLASMAS IN SPACE (AMPS) SPACELAB PAYLOAD DEFINITION STUDY

Prepared for

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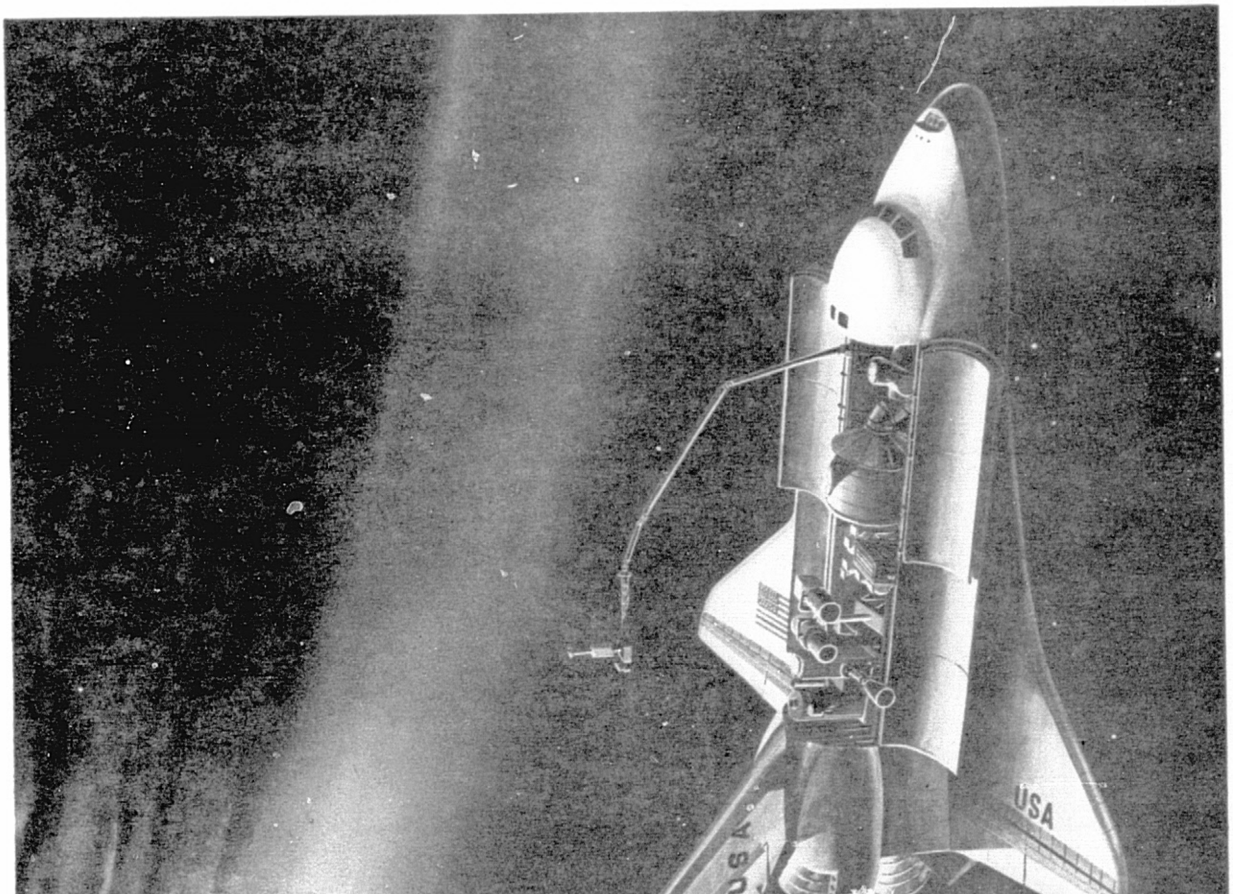

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FOREWORD

The AMPS Technical Summary Document is submitted by Martin Marietta in accordance with Data Procurement Document Number 486, Revision A, of Goddard Space Flight Center NAS8-31689.

TABLE OF CONTENTS

TABLE OF CONTENTS (Continued)

TABLE OF CONTENTS (Continued)

TABLE OF CONTENTS (Continued)

LIST OF FIGURES

Page

Page

Page

Page

Page

Page

		Page
Table 3.2-1	Flight 1 Science Requirements	3-11
Table 3.2-2	Flight 2 Science Requirements	3-12
Table 3.3.2-1	Crew Training Time	3-35
Table 3.4-1	Mission Scientific Objectives/Experiment Summary	3-37
Table 3.4.1-1	Flight 1 Experiment/Instrument Summary . . .	3-39
Table 3.4.2-1	Flight 2 Experiment/Instrument Summary . . .	3-48
Table 3.5.1-1	Flight 1 Experiment/Instrument/Support Equipment Summary	3-58
Table 3.5.2-1	Flight 2 Experiment/Instrument/Support Equipment Summary	3-65
Table 3.5.3-1	Follow-on Flight Configuration Impacts	3-67
Table 4.1-1	Structures/Mechanisms Requirements as Derived from Science Requirements	4-2
Table 4.1-2	Comparison of Structural Mounting Concepts . .	4-5
Table 4.2-1	Shuttle/Spacelab Thermal Control Capabilities	4-15
Table 4.2-2	Thermal Analysis Groundrules, Guidelines and Assumptions	4-19
Table 4.2-3	AMPS Flight 1 Thermal Performance Summary. . .	4-26
Table 4.2-4	AMPS Flight 1 Heater Power	4-27
Table 4.3-1	High Voltage Power Supply Requirements	4-36
Table 4.3-2	Flight 1 Energy Usage Summary.	4-39
Table 4.4-1	Flight 1 Pointing Requirements	4-42
Table 4.4-2	Flight 2 Pointing Requirements	4-43
Table 4.4-3	Orbiter/Spacelab Pointing Accuracy Capability.	4-44

		Page
Table 4.5-1	Flight 1 Data and Command Requirements . .	4-55
Table 4.5-2	Flight 2 Data and Command Requirements . .	4-56
Table 4.5-3	Flight 1 and 2 DMS Comparison.	4-69
Table 4.5-4	RAU Allocation Comparison.	4-74
Table 4.5-5	Detached Payload Versus Orbiter Capability.	4-75
Table 4.5-6	Constant Bandwidth Channel Characteristics	4-77
Table 4.6-1	Flight 1 C&D Functional Requirements Summary	4-79
Table 4.6-2	Flight 2 C&D Functional Requirements Summary	4-80
Table 4.6-3	Payload Specialist/Ground Attributes . . .	4-86
Table 4.6-4	Operations Allocation.	4-87
Table 4.7-1	AMPS Communication Requirements	4-93
Table 4.8-1	Environmental Sensing Package Weight Breakdown	4-107
Table 4.8-2	Beam Diagnostic Package Weight Breakdown .	4-109
Table 4.8-3	RF Receiver Package Weight Breakdown . . .	4-112
Table 4.8-4	Gas Release Module Weight Breakdown . . .	4-115
Table 4.8-5	Chemical Release Weight Breakdown.	4-118
Table 4.8-6	Instruments for the Plasma Flow Experiment	4-119
Table 4.8-7	Plasma Flow Diagnostics Package Weight Breakdown	4-121
Table 4.9-1	AMPS Mission Planning Software	4-125
Table 4.9-2	AMPS Payload Integration, Test and Launch Software	4-126

1. INTRODUCTION

This document records the results of the AMPS Phase B study which was initiated November 4, 1975.

The Atmosphere, Magnetosphere and Plasmas in Space (AMPS) Spacelab Payload program uses the capability of the Space Transportation System (STS) to provide an orbital national research laboratory for scientific investigations in these areas. The laboratory uses coordinated instrument groups, complemented by flight and ground support equipment, to study the earth's near-space environment and solar/terrestrial physics.

Investigations are conducted either by using controlled, active stimulation experiments in conjunction with remote and in situ sensing and diagnostic instruments or by using stimulation experiments in a strictly passive mode. The laboratory uses instruments built for NASA by various American universities and contractors, as well as by space research groups in other countries. Foreign researchers also participate by proposing investigations. The return-from-orbit capability provided by the Space Transportation System (STS) introduces a new era that allows multiple reuse of the instruments and extension of the instruments' capabilities during the life of the program. This permits the development of an evolutionary science program that is both economical and responsive to changing requirements.

The study was initiated with the evaluation of some 60 instrument candidates and 80 possible science investigations. The early analysis emphasized the science aspect in terms of the functional requirements for each of the potential experiments identified by the AMPS science working group. These requirements were then used for the grouping of instruments into practical payloads which would fit the capabilities of the Shuttle/Spacelab. This analysis, performed during the pre-proposal and early Phase B Study period, resulted in the definition of eleven different AMPS configurations. The data was then used to define a typical set of requirements for a flexible AMPS laboratory. The Requirements Review, held in January of 1976, outlined these requirements and addressed the limitations imposed by use of the STS. The data gathered to this point in the study showed that a planned sequential buildup c' the laboratory would be necessary to meet both physical and funding limitations. This led to the definition of five strawman payloads by the science working group, which were used to establish a conceptual laboratory and to define preliminary design of a configuration which could satisfy AMPS needs during the early program period.

2. SUMMARY

This document has been assembled to report the results of the AMPS Phase B study at three major levels. The first part summarizes the program, mission, operations, experiment/instrument and overall system configuration definitions which have been developed by the study. The second part describes each of the subsystem approaches which have been selected to support the science payloads. The third part presents the supporting analyses (both systems and subsystems level) accomplished to assure that the configurations are capable of satisfactory performance.

The analysis and developed configurations were based on implementing the five strawman payloads defined by the AMPS science working group. The major goal was to use these payloads to develop an AMPS laboratory capable of initially accomplishing significant science investigations and which could be evolved over a period of time to a sophisticated laboratory to do the complete series of investigations foreseen for the future. Specific emphasis was placed on a detailed configuration for the first of the strawman flights. A second flight configuration was then developed around the premise of maximum use of Flight 1 designs to accomplish the science investigations desired. A top-level evaluation of unique requirements for the balance of the five payloads was completed in order to define the design drivers and potential changes as the laboratory evolved.

The results of these analyses have shown that the concept of an evolving AMPS laboratory is viable and that it is adaptable to a changing instrument complement. There are, of course, limitations such as total payload carrying capability, payload center of gravity requirements, available field-of-view from the Orbiter payload bay, funding levels, etc. These limitations can be overcome by effective planning, use of existing designs or equipment, maximum use of STS capabilities and careful attention to payload integration.

Significant Issues - Although Phase B study results indicate the feasibility of development of a laboratory to accomplish AMPS experimentation, several significant issues were highlighted. Most of these areas not only impact implementation of the AMPS program but are directly applicable to any scientific payload to be flown on Spacelab. These issues should be addressed early in the process of future planning for specific Labcraft payloads and are summarized below:

(1) Integration design for mounting instruments and Labcraft equipment on Spacelab pallets cannot be accomplished without detailed stiffness and dynamic characteristics of the ESA supplied pallets. This data is of particular importance in the determination of the bending characteristics between pallets and their impact on pointing platform stability and accuracy performance. It is also required to develop structural design criteria for instrument support

trusses and brackets based on load transmission from the Orbiter through the pallets to the payload. Emphasis should be placed on acquiring detailed structural models from ESA.

(2) Instrument pointing requirements and weights vary considerably for the AMPS payload and will continue to do so for many of the Spacelab payloads. The configuration described in this report assumed that two types of pointing platforms would be available in the NASA inventory, in addition to the ESA supplied IPS for large instruments (500 to 2000 Kg). AMPS investigations, requiring multiple, independent, simultaneous pointing, can best be accommodated by two types of platforms: center of mass and end mounted. Quantity of targets and available space on the pallet were the prime drivers. A cursory evaluation of future payload pointing requirements coupled with the AMPS analysis lead to the suggestion that various sizes of pointing platforms be supplied for the Shuttle payload era.

(3) The analysis presented in this report was based on 7-day missions. Mission time can be increased, but will be limited by the penalty for additional commodities necessary to support the longer mission duration. Careful consideration should be given, during the initial phase of future payload acquisition, to prioritizing investigations so that the flexibility exists to extend the mission when time of experimentation is a significant parameter.

(4) The effect of spacecraft generated environments on instrument performance has not been established. Expected EMI levels, as well as contaminants produced by the spacecraft and instruments, indicate that serious degradation may be imposed on many of the more sensitive instruments . The effects of spacecraft charging on the operation of instruments which are programmed to perturb the surrounding plasma could possibly call for redesign of some of the investigations. Presently defined studies and analyses should be enhanced to determine actual instrument susceptibility and develop work around methods and design criteria for both instruments and spacecraft.

(5) The complex scientific nature of the varied Spacelab payload missions, together with the inherent need for close coordination with investigators, dictates the provision of a dedicated Payload Operations Control Center (POCC). Although requirements for such a center have been addressed for specific payloads, the need exists to develop a detailed set of requirements which will define the overall center. This definition should be based on the requirements for several different payloads and should stress development of the requirements to a level which will size the facility and

define the necessary support equipment. Consideration should be given to: real time display requirements, computational capability, interfaces with science investigators, flexibility to accommodate a variety of missions and optimizing the mix of ground versus onboard control.

(6) The time allotted for payload oriented activities during Level III, II and I integration and checkout at KSC is necessarily limited in order to accommodate the quantity of payloads programmed for the STS. Evaluations to date indicate that preintegration of the total payload is required prior to entering the KSC flow with Spacelab and Orbiter. Definition of a dedicated Payload Handling Facility (PHF) should be undertaken, based on the requirements of several payloads. This definition should emphasize the use of existing KSC facilities and develop the necessary support equipment requirements. Consideration should be given to: specific GSE to support payload checkout, use of GSE supplied for the O&C building through a data link and simultaneous payload operations.

3. PAYLOAD DEFINITION AND PRELIMINARY DESIGN

3.1 Program Definition

The overall goal of the AMPS program is to develop a laboratory, using Orbiter and Spacelab provided capability, to perform a wide range of investigations leading to the understanding of the physical processes that control man's near-earth environment. The first step in the program definition process is to develop an understanding of the scientific phenomenon which is subject to investigation and to define the overall science objectives of the AMPS program.

3.1.1 Science Objectives

AMPS studies emphasize the importance of understanding the underlying physical processes that transfer and distribute the energy, mass, and momentum from the sun to the earth's magnetospheric/atmospheric system. The broad scientific goals generated for AMPS provide timely data and recommendations as to the problems that affect both man's practical and intellectual needs. The goals of these studies are:

(1) To improve our understanding of natural and man-made disturbances in the magnetosphere and atmosphere that disrupt the everyday activities of terrestrial life.

(2) To investigate the universal physical processes that control phenomena fundamental to the basic nature and characteristics of our universe.

(3) To use the AMPS laboratory as a research tool to build on, complement, and extend the solid foundations laid by previous programs including the International Magnetosphere Study, Electrodynamics Explorer, Atmospheric Explorer, Nimbus, and the Solar Maximum Mission.

The environment to be investigated includes the atmosphere, the ionosphere, the magnetosphere, and the solar wind interacting in a dynamic mode. These regions are influenced by processes originating near the surface of the earth. The extent to which the elements of the earth's environmental system are interdependent continues to be dramatically emphasized through the growing concern over the effects of both natural and man-made catalytic agents such as nitrogen oxides and chlorine compounds on the concentrations of naturally occurring stratospheric ozone.

The solar wind/magnetosphere/ionosphere/atmosphere system is characterized by several physical mechanisms that govern its behavior as parts of complex closed chains of cause and effect relationships, as shown in Figure 3.1.1-1. The large-scale dynamic

processes that occur in these regions include a wide range of spatial and temporal scales and are inherently so complex and non-linear that to gain an understanding of the basic processes requires application of a wide variety of experimental techniques, including optical remote sensing, multiple-satellite in situ diagnostics, and newly developed active perturbation experiments.

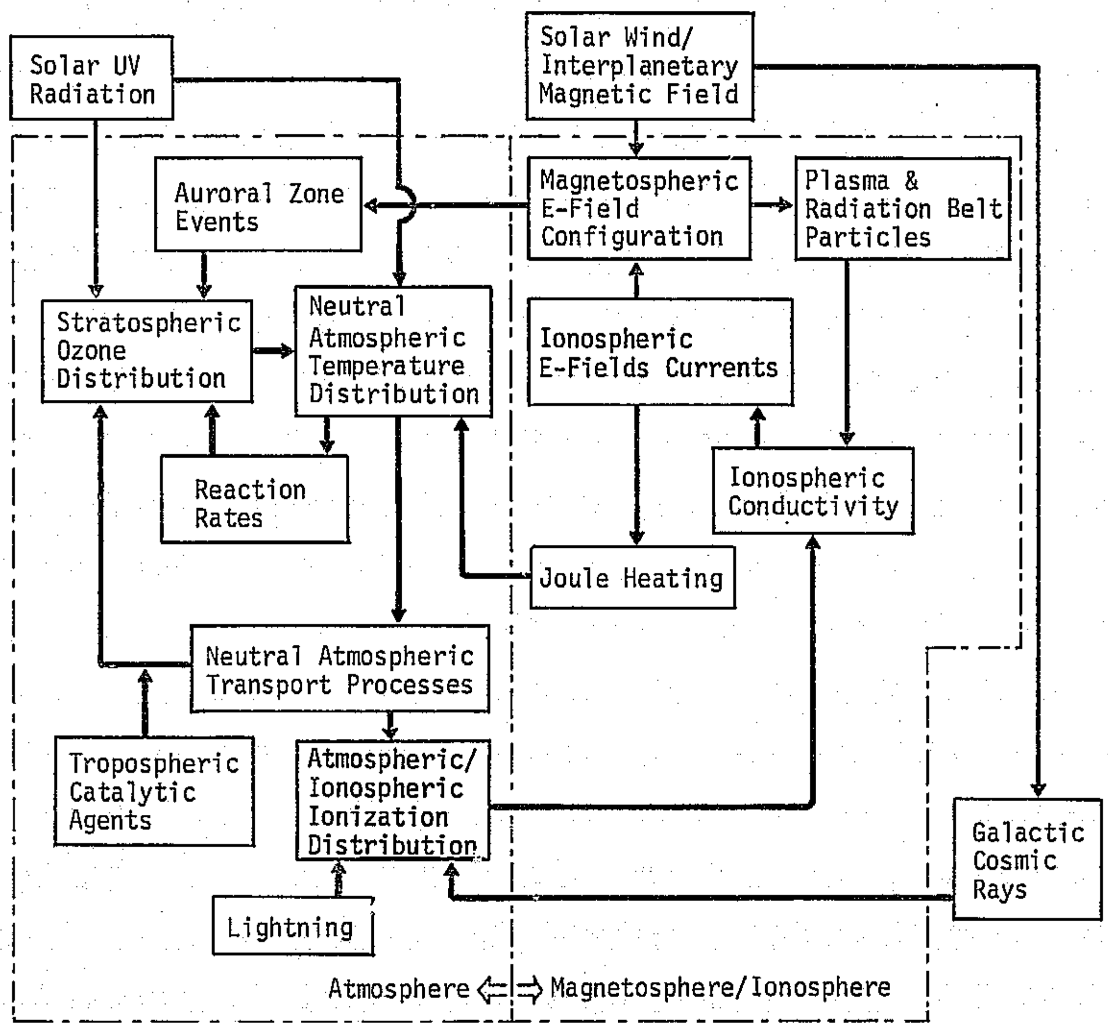


Figure 3.1.1-1 Interactive Scheme for Solar Driven Magnetospheric/Atmospheric Coupling

Atmospheric Physics - The chemistry, dynamics, and energetics of the stratosphere, mesosphere, and lower thermosphere are particularly important because most solar radiation is deposited here and influences the general terrestrial climate. As shown in Figure 3.1.1-2 the relative importance of various energy sources such as solar radiation, tropospheric wave energy, and joule and energetic particle heating from the magnetosphere to the structure of the atmosphere must be established. The rate of input of natural and man-made chemical species such as freon to the troposphere is believed to have a significant control over the stratospheric composition. The relationship between minor constituent photochemistry and the energetics and dynamics of the stratosphere and mesosphere is essential to AMPS investigations and the potential consequences of these interactions warrant careful study.

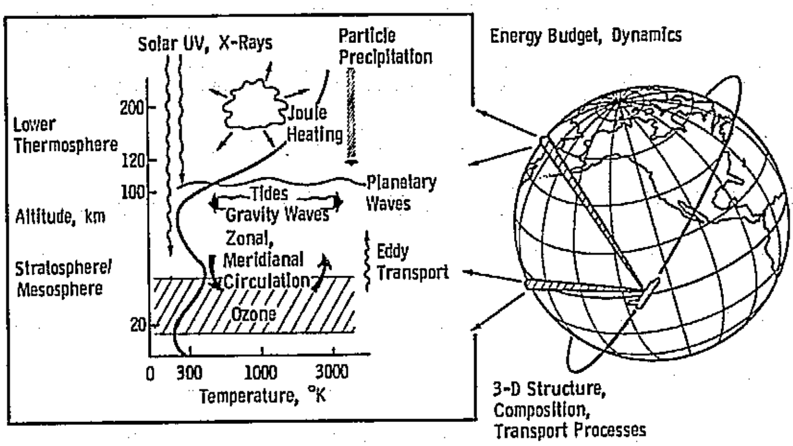


Figure 3.1.1-2 AMPS Atmospheric Science Studies

Magnetospheric Physics - Magnetospheric physics emphasizes studies of the physical processes that couple the solar wind, the magnetosphere and the atmosphere. AMPS scientific investigations need to address the interchange of mass, momentum, and energy between the magnetosphere, the ionosphere, and the atmosphere. The fraction of the incident solar-wind energy captured by the magnetosphere may vary more than an order of magnitude. The variety of paths through which this energy is conducted to the atmosphere involve direct particle injections, magnetic energy storage, particle acceleration, electromagnetic waves, and joule heating of the ionosphere.

The bulk motion of plasma and their associated electric fields, and the wide variety of plasma instabilities involved in major magnetospheric processes are included. Plasma instabilities may

cause anomalous resistivity that restricts field-aligned current flow that in turn evolves field-aligned electric fields and contributes to particle acceleration to enhance or modulate precipitation into the atmosphere as indicated in Figure 3.1.1-3. Particle precipitation produces auroras and ionization that modify the ionospheric conductivity and alter the general electric field and current distribution. Study of the fundamental physical processes that couple these regimes of the earth's environment is a major area for AMPS investigations.

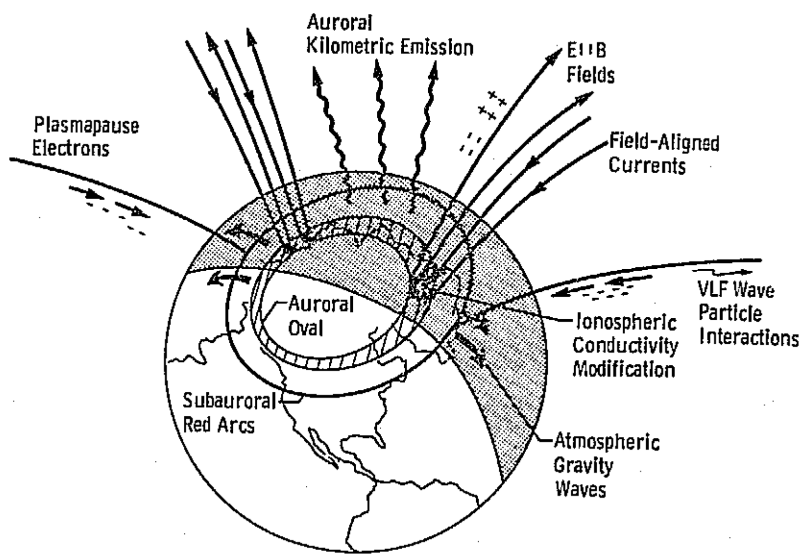


Figure 3.1.1-3 AMPS Magnetospheric Studies

Plasma Physics - AMPS will also use the ionosphere as a plasma laboratory, in addition to the study of the natural environment. The bulk of matter in the universe is in the plasma state--ranging from dense collision-dominated ionized gases to tenuous collision-free plasma, including stellar atmospheres, planetary ionospheres and magnetospheres, the solar wind, and the interplanetary, interstellar, and intergalactic media. The large-scale, nearly homogeneous, ionospheric plasma is an ideal medium in which scientists can examine the fundamental astrophysical processes involving basic plasma flow, beam-plasma interaction, wave/particle, and wave/wave interactions, that are difficult to conduct in the ground-based laboratory, as illustrated in Figure 3.1.1-4.

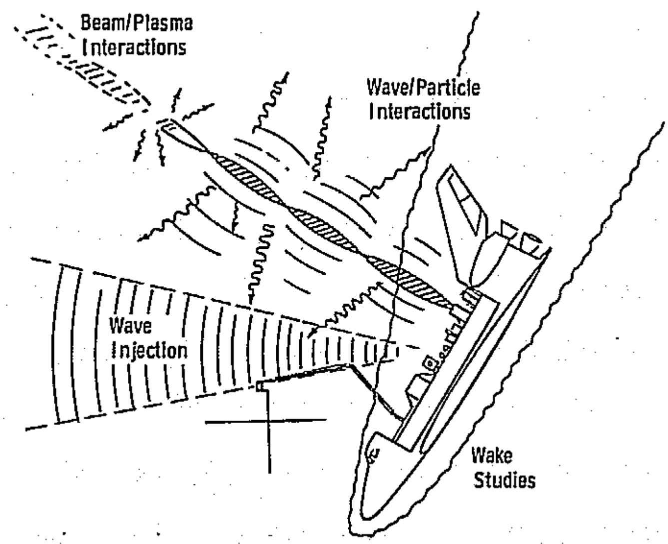


Figure 3.1.1-4 AMPS Plasma Physics Studies

3.1.2 Program Objectives

The second step was to develop program objectives based on maximum satisfaction of the science requirements but within overall program funding and STS payload limitations. These program objectives are:

(1) The laboratory design should be flexible in order to accomplish the science objectives to the greatest extent possible. The initial configuration should be based on providing a significant investigative capability for minimum cost.

(2) The laboratory design should emphasize adaptability to evolve as the program progresses by addition and minimum modification of the support equipment required by the instruments. Cost spreading, over the total program period, will be a major consideration in the evolution process.

(3) The laboratory should be designed to carry a variety of instrument complements. Changes in the types of experiments due to new scientific data as well as delays in development of beyond the state-of-the-art instruments must be accommodated with a minimum of changes to the laboratory.

(4) The developed configurations should emphasize a building block approach to fit a variety of potential funding limitations.

(5) Modular design features should be considered in order to accommodate integration of complete experiments into multi-discipline Spacelab payloads.

(6) The laboratory should be designed for rapid turnaround in order to provide a yearly multi-mission capability. Design for ease of refurbishment and modification will be a major goal.

3.1.3 Program Planning

Preliminary planning was initiated as the third step in program definition. Early evaluations established that the Orbiter would carry a standard Spacelab configuration of a tunnel, short pressurized module, three pallets and subsystem equipment to support the science payloads under consideration. The two flights defined were to be launched in the 1981 and 1982 timeframe and would be conducted over a seven day period. GSFC has been assigned as mission manager and support will be provided by instrument developers (Universities and contractors) and a prime contractor for design and development of AMPS unique support equipment and payload integration. Early in the Phase B study guidelines and assumptions, as outlined below, were defined so as to direct configuration definition and analyses to fit the program needs.

- Orbiter will provide transportation to orbit
- Spacelab pressurized module, three pallets, and subsystem support equipment will be provided as GFE.
- Maximum use shall be made of Orbiter/Spacelab capabilities
- Labcraft equipment, designed for multi-mission usage shall be used to the maximum extent possible.
- Off-the-shelf equipment (within safety requirements) shall be used wherever possible.
- The AMPS laboratory will be designed to evolve over a long range program.
- Scientific instruments will be provided as GFE.
- Costs will be estimated for two flights.

- o Detailed evaluation will be accomplished for the first two strawman payloads developed by the AMPS Science Working Group.
- o The design will accommodate a maximum of five reflights per year (no configuration change) or two major configurations per year.
- o Launches will be conducted from ETR.
- o Satisfaction of all strawman payload science requirements at minimum cost shall be a prime design goal.
- o The prime contractor shall provide:
  - AMPS unique instrument flight support equipment not provided by Orbiter/Spacelab
  - AMPS unique GSE
  - Interface design and hardware
  - Payload integration and checkout for two flights
  - System software development and instrument software support
  - Level IV Integration at Contractor's plant and KSC
  - Assurance of payload/STS compatibility
- o Prime contractor shall support:
  - Level III, II, and I integration at KSC
  - Instrument development in terms of consultation and interface management
  - Flight operations during the missions
- o Schedules used for planning purposes are shown in Figure 3.1.3-1.

3.1.4 Phase C/D Effort

The fourth step was to evaluate the program starting with initial instrument definition and continuing through launch of the payload. This analysis resulted in the definition of the tasks necessary to develop and fly the laboratory and supported estimates of program costs. A summary of the Phase C/D tasks and their interrelationship is presented in Figure 3.1.4-1. The program is envisioned in four parts: instrument development and certification; flight support

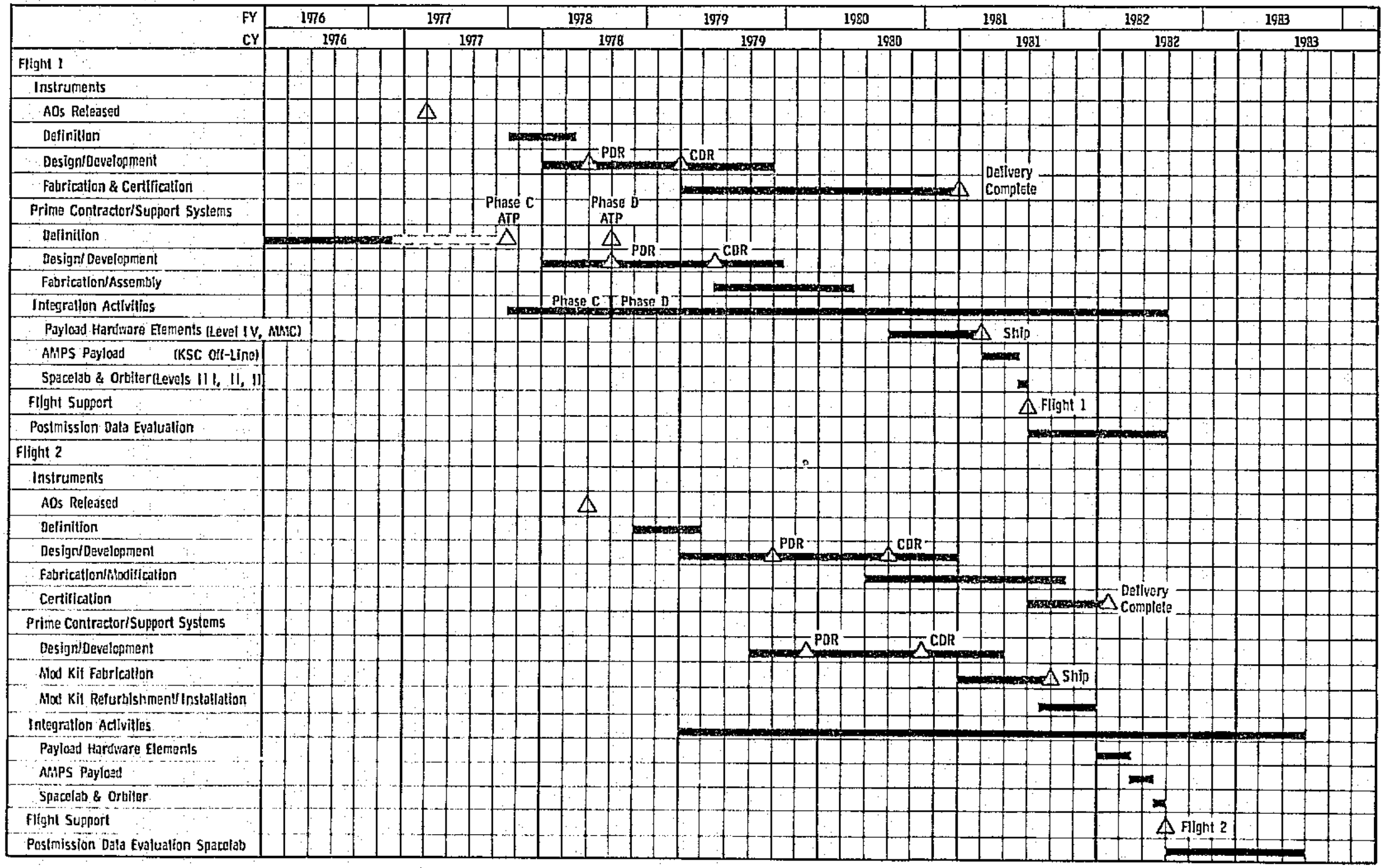


Figure 3.1.3-1 Program Schedule

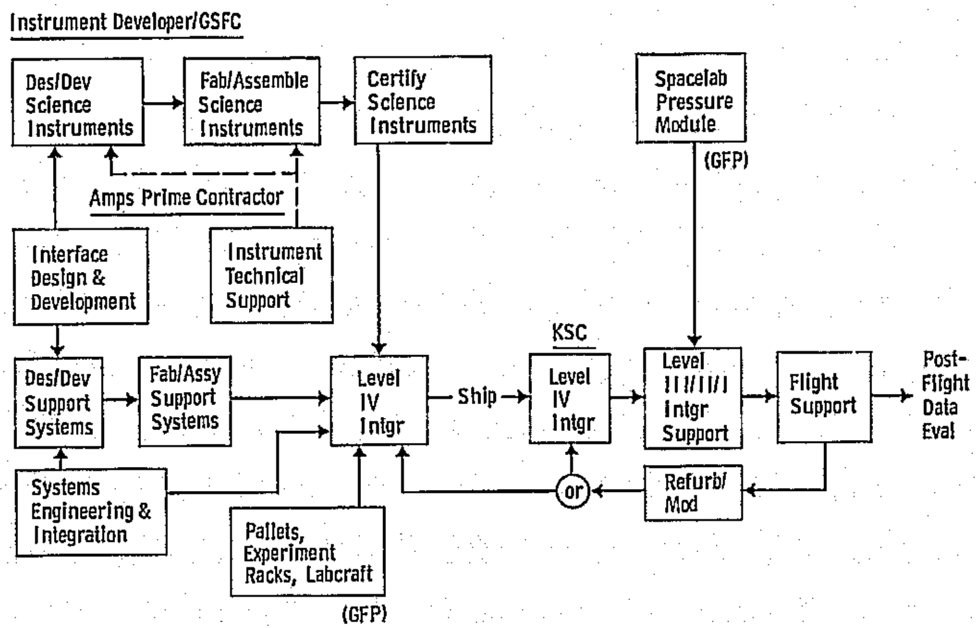


Figure 3.1.4-1 Major Phase C/D Tasks

equipment and interface design, development and certification; integration of the AMPS equipment with the Orbiter/Spacelab provided equipment; and support of ground and flight operations. The definition of these tasks and the approach to overall program management and program cost estimates are contained in documents MA-04 (Program Analysis and Planning for Phase C/D) and MF003R (Program Study Cost Estimates) respectively.

3.2 Mission Definition

Definition for the AMPS mission emphasized maximum accomplishment of the science and program objectives as stated in the previous section. The analysis was initiated by first evaluating each of the Flight 1 and 2 investigations defined by the science working group in terms of specific requirements which have an impact on mission performance.

Table 3.2-1 summarizes the mission requirements imposed by the science program defined for Flight 1. Comparison of these requirements with the various scientific tasks leads to the identification of drivers in the development of mission parameters. For example, the maximum operating range of the LIDAR, for minor constituent mapping, and the optimum ground to gas release distance define the preferred altitude. The ground station position with respect to the gas release as well as the capability to map the minor constituents of the atmosphere over selected areas of the earth drive the orbital inclination selected. The desire to map as much of the earth's atmosphere as practical during a seven-day mission was a prime driver in the time allowed for the measurement of minor constituents. Time of day, time of year and specified target requirements impact experiment scheduling. Specific science targets also impact Orbiter orientation. The location of available ground stations, with respect to the orbit, dictate the scheduling of the gas release experiment. Other investigations, such as Orbiter wake mapping, beam diagnostics and EMI measurements, solar flux and contamination monitoring do not impose limiting requirements and therefore can be easily integrated into the mission. Maximum use of deployed instrumentation (by RMS and free flying) prior to release for detached operation suggest scheduling later in the mission.

Flight 2 requirements, as summarized in Table 3.2-2 are similar to those for Flight 1. For example, the minor constituents mapping investigations impacted this flight to the same extent as Flight 1. The available ground station location with respect to the orbit dictated the scheduling of the conductivity modification experiment as well as the mission altitude selection. The wave particle interaction, long delay echo, plasma flow and solar flux monitoring have less limiting requirements and therefore could be scheduled between other investigations.

The second portion of the mission definition was to integrate the crew into the experiment scheduling. The normal, non-experiment oriented activities for the crew were defined and an example of a typical day is portrayed in Figure 3.2-1. The amount and types of experimentation as well as the need for simultaneous operation of instruments and support equipment led to sizing the crew at two payload specialists and two mission specialists. The payload specialist is responsible for payload operations and collecting the scientific data where the mission specialist operates support equipment and supports the payload specialist during experiment operations. Two twelve-hour shifts were considered

Table 3.2-1 Flight 1 Science Requirements

Attribute / Scientific Task	Orbital Condition				Repetition/Frequency		Target					Attitude			Support		Remarks
	Orbiter Altitude, km	Inclination	Time of Day	Time of Year	Number/Mission	Frequency	Sun	Moon	Limb	Nadir	Magnetic Field Line	+ZLV, XPOP	Solar Inertial, XPOP	Other	Ground Station	Independent Satellite	Note Number
Minor Constituents	200-250	High As Possible		1 Each Season		Near Continuous	X		X	X		X				X	
Acoustic Gravity Waves	200-300		Terminator		6-7	1/Day						X		X	X		
Orbiter Wake Mapping						1/Mission								X			1
Beam Diagnostics First-Generation Electron Accelerator			Night		3	1/Day					X			X			2
Solar Flux Monitor			Day		6	1/Day	X						X				
EMI Measurement/ Mapping						Map Each Configuration											3
Contamination Monitoring						Continuous Mapping											

Note:
1. Vehicle attitude is XPOP, Z in orbital plane, and roll around the X axis.
2. Electron accelerator is hard-mounted with beam transmitted along Z axis. Therefore, Z axis must be pointed within 10 degrees of the local magnetic field.
3. Part of ESP.

Table 3.2-2 Flight 2 Science Requirements

Attribute / Scientific Task	Orbital Condition: Orbiter Altitude, km	Inclination	Time of Day	Time of Year	Repetition/Frequency: Number/Mission	Frequency	Target: Sun	Moon	Limb	Nadir	Magnetic Field Line	Attitude: +ZLV, XPOP	Solar Inertial, XPOP	Other	Support: Ground Station	Independent Satellite	Remarks: Note Number
Minor Constituents	200-250	High as Possible		1 Each Season		Near Continuous	X		X	X		X				X	
Conductivity Modification	150-200	59 degrees	Terminator		1							X			X		1
Wave/Particle Interaction A					3	1/Day											
Wave/Particle Interaction B			High Latitude		1												3
Long-Delay Echo			High Latitude		1												2
Plasma Flow			Two Complete Revolutions		1									X			4
Solar Flux Monitor			Day		6	1/Day	X						X				

Note: 1. Ft. Churchill and Port Nelson are ground stations.
2. Requires subsatellite approximately 80 km behind orbiter.
3. Run in conjunction with long-delay echo.
4. Requires deployed wake generator.

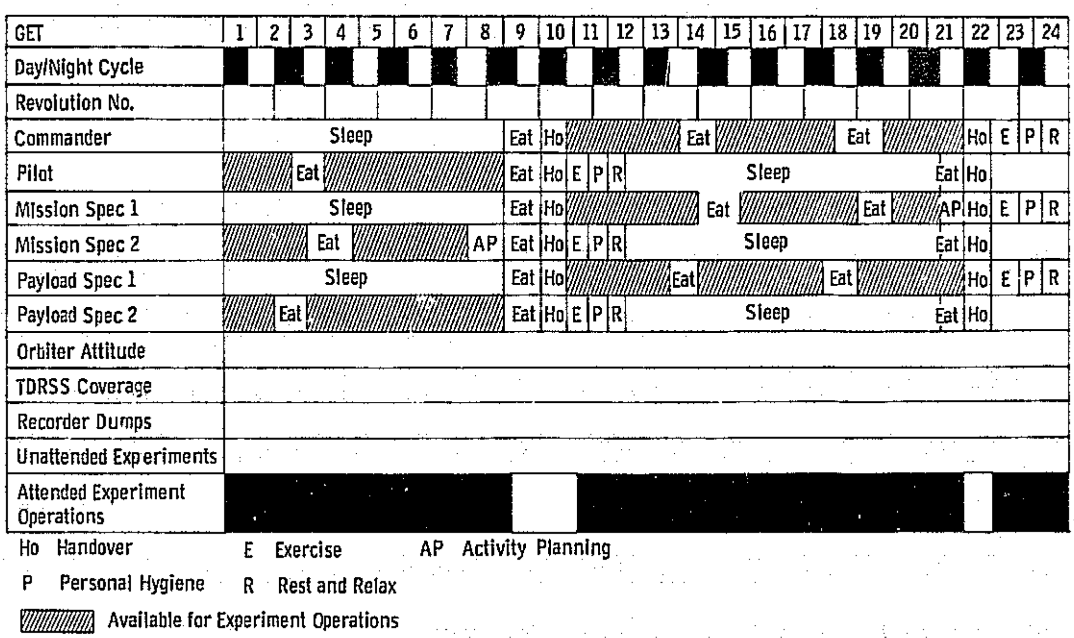


Figure 3.2-1 Typical Crew Day (non-experiment activities)

in the planning. The physiological requirements of the flight crew that were considered include: (1) each crewman must receive eight hours of sleep in a 24-hour period, (2) the start time for the sleep cycle should occur ± 1 hour of the same time each day, (3) crewmen on each shift have simultaneous sleep cycles, (4) exercise, rest and relaxation, and personal hygiene periods must be provided, and (5) handover and activity planning periods are required for smooth mission operations. These crew scheduling constraints, reasonable meal time periods, the mission science requirements, and the ground support requirements were interleaved to form the integrated mission timeline as depicted in Figure 3.2-2. This figure presents a single day for Flight 1 as an example; and the balance of the mission timeline can be found in the Final Review presentation material.

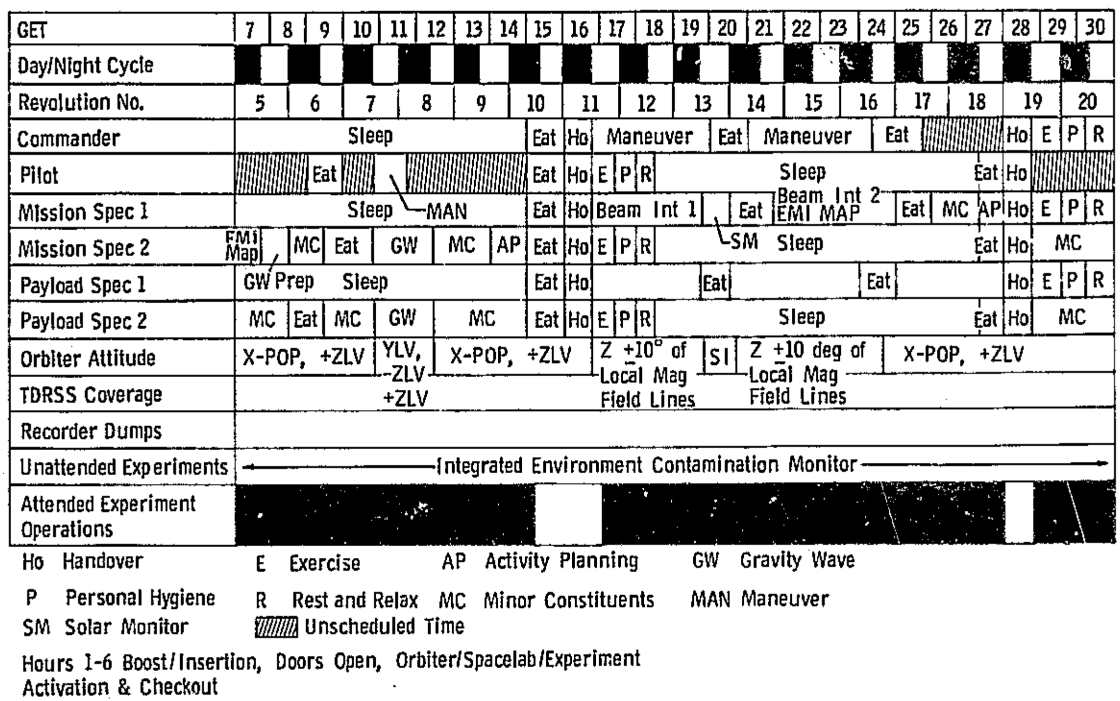


Figure 3.2-2 Flight 1 Day 1 Integrated Mission Timeline

The third step in planning the mission was to address instruments required for each investigation and develop a feasible experiment schedule. Figure 3.2-3 depicts the results of this analysis for a typical day and additional timelines were included in the Final Review presentation material. This data was also used to define power and data profiles leading to subsystem preliminary design. This process was iterated sufficiently to design viable missions for Flight 1 and 2 as defined in the following paragraphs.

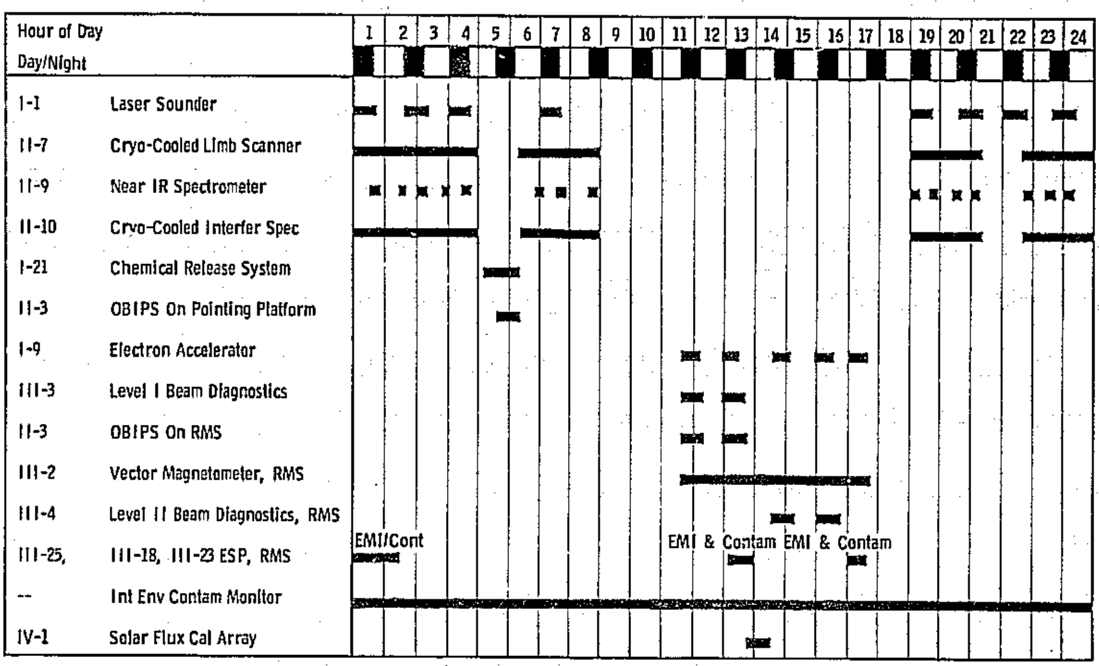


Figure 3.2-3 Instrument Timeline, Day 1 Flight 1

The combination of objectives, mission requirements and STS limitations led to the definition of a series of parameters around which the specific AMPS Flight 1 and 2 missions were defined. These parameters are outlined below.

Altitude	:	200-210 Km
Inclination	:	57^{o} (as high as possible)
Launch Site	:	ETR
Duration	:	7 days (extension possible with reduced payload)

Crew	:	Pilot, Commander, 2 Mission Specialists, 2 Payload Specialists
Maximum Payload Weight	:	32,000 lb (Landed)
Payload	:	Instruments as noted in paragraph 3.4 (No more than 1 experiment at a time)
Orientation	:	Varied to fit experiment requirements

<u>Flight 1 Mission Description</u> - Figure 3.2-4 portrays the Flight 1 profile in terms of activation, deactivation, and experiment operation scheduling. The times were estimated and include setup and shutdown. The minor constituent experiment requires four instruments: laser sounder, cryogenically cooled IR interferometer/spectrometer, cryogenically cooled limb scanner, and near IR spectrometer. The Orbiter is oriented with the X axis perpendicular to the orbit plane and the Z axis pointed at Nadir. The laser sounder is then operated through a range of instrument generated conditions to collect data. Simultaneously, the two cryo cooled instruments are pointed at the limb and controlled via a pointing platform to perform remote measurements of specific atmospheric constituents. The near IR spectrometer is also pointed toward the limb via a separate pointing platform and performs occultation measurements of the sun at both terminators. The capability is provided to point either with or against the velocity vector. This experiment is repeated for a significant portion of the flight as shown in the profile.

Elements	Flight Schedule (Day 1, Day 2, Day 3, Day 4, Day 5, Day 6, Day 7)	Total, hr
Ascent & Operational Preparation		7
On-Orbit Experiment Operations		144
Minor Constituents		93
Acoustic Gravity Waves		12
Orbiter Wake Mapping	Coordinate Near-Orbiter Measurements with Orbiter Maneuvers	10
Beam Diagnostic Measurements		
Level 1		9
Level 2		9
Solar Flux Monitoring		5
EMI Measurement/Mapping	Simultaneous with Various Experiments	6
Contamination Monitoring		144
Deactivation & Thermal Conditioning	Dependent on Power Availability	16
Reenty & Landing		1

Note: All times are approximate and intended for overall scheduling assessment.

Figure 3.2-4 Flight I Profile

The acoustic gravity wave experiment is conducted six times during the flight. The mission has been designed to provide the proper orbital position with respect to the observing ground station and for specific lighting conditions. The gas release module is ejected from its pallet location in the direction of the velocity vector. The natural trajectory, based on approximately 5 meters per second delta velocity, will carry the module to the proper position in approximately one orbit where the release is commanded. The path of the module is determined by tracking via the Orbiter provided rendezvous radar to determine the exact time for release. Observations of the formed gas cloud will be performed from the ground station and onboard the Spacelab using a gimballed low-light level TV camera.

The Orbiter wake mapping experiment is performed using a series of instruments packaged as an integrated module (Environmental Sensing Package, ESP). Two modes of operation are envisioned. The ESP will be extended perpendicular to the pallet by means of the RMS, spun up at 4 RPM, and the Orbiter will roll about the X axis so that the instruments perform measurements in the ambient and Orbiter generated wake. The second mode of operation calls for ejecting the ESP (spinning at 4 RPM for stability purposes) from the RMS. The ejection is along the velocity vector and the natural trajectory will carry the ESP through the far Orbiter wake approximately three times. Continuous tracking of the ESP will be accomplished via the Orbiter rendezvous radar so as to obtain relative positioning data. The ESP will be commandable from and will send data back through an AMPS provided communication link.

The beam diagnostic measurements experiment requires several instruments: electron accelerator, gas plume release, beam diagnostic group, low light level TV camera and magnetometer, which are packaged as an integrated module. Two modes of experimentation are envisioned. The first mode requires Orbiter orientation of the electron accelerator axis along the magnetic field lines, pulsing the accelerator, releasing a gas into the beam and viewing the beam structure with a low light level TV camera deployed into position by the RMS. For the second mode, the orientation remains the same and the electron accelerator is pulsed after positioning the diagnostics instruments in the path of the beam. Beam characteristic data will be transmitted by the AMPS communication link. Both modes are repeated three times during the flight.

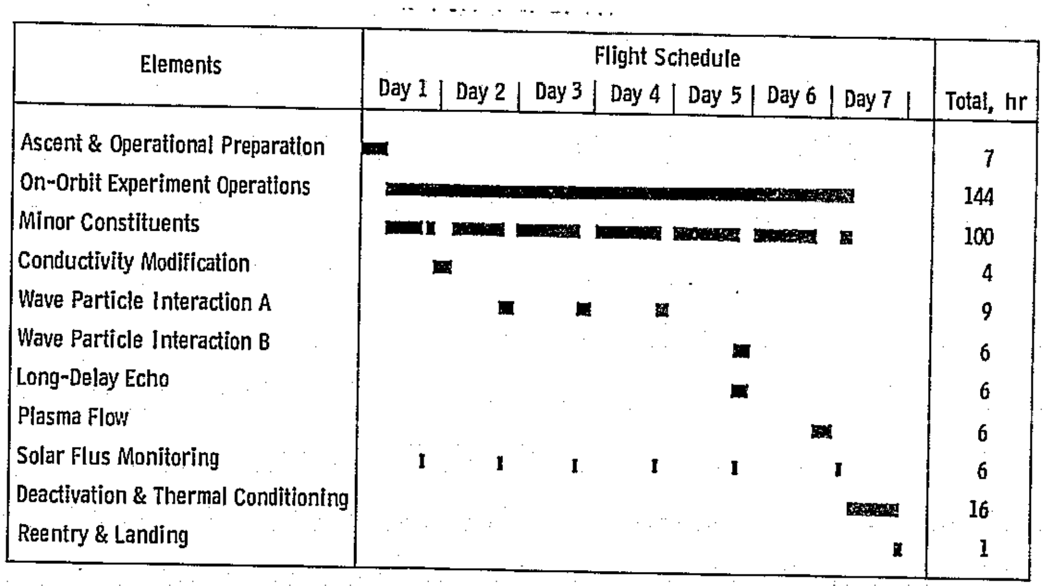
The solar flux monitoring experiment is repeated six times during the flight. The Orbiter will orient the axis of the instrument toward the sun just prior to sunrise and operate in a free drift mode while measurements are in process. This data will be used to calibrate free-flying satellite measurements taken for other programs.

The EMI measurement/mapping experiment is performed using a series of instruments packaged in the ESP (also used for wake mapping). Two modes of experimentation are envisioned. The ESP will be deployed on the RMS and moved to various positions over the Orbiter payload bay.

Sufficient measurements will be made to map the electrostatic and magnetic fields within the range of the RMS. Various payload operating conditions will be investigated by performing several of the AMPS experiments while monitoring EMI levels. The second mode requires ejection of the ESP to measure the far Orbiter fields until the levels become insignificant. Continuous tracking of the ESP will provide location data with respect to the Orbiter.

The contamination monitoring experiment consists of operating a self-contained series of instruments (data is recorded and returned for evaluation on the ground). This package measures various contamination parameters within and around the Orbiter payload bay from a fixed location on the aft pallet. Operation is envisioned as continuous within the limitations of electrical energy available.

<u>Flight 2 Mission Description</u> - Figure 3.2-5 portrays the Flight 2 profile. The minor constituent experiment performance is nearly identical to Flight 1 except for the addition of two UV-VIS-NIR spectrometer/photometer instruments which are packaged with the near IR spectrometer and pointed at specific targets on the earth or at the limb and controlled by the same pointing platform used in Flight 1. The data taking from these new instruments is scheduled around the NIR spectrometer occultation investigation. The Solar Flux monitoring experiment is identical to Flight 1.



Note: All times are approximate and intended for overall scheduling assessment.

Figure 3.2-5 Flight 2 Profile

The conductivity modification experiment is conducted once during the flight. The mission has again been designed to position the chemical release module to the proper orbital location with respect to the available ground observation station. The module is removed from the pallet by the RMS and the Orbiter flies the laboratory to the proper relative position for monitoring the release cloud (approximately 80 Km). Tracking of the module will be accomplished in order to determine the exact release time. The release cloud is then observed from the ground station or from the laboratory by the low light level TV and UV-NIR-VIS spectrometer/photometer at the time of release and on the succeeding orbit.

The wave particle interaction experiments require an RF transmitter, antenna and receiver mounted on the pallet and an RF receiver and antenna contained in a free-flying module. The first mode requires positioning of the fixed antenna, by the Orbiter, to various orientations with respect to the earth's magnetic field lines. The instrument will then be operated in a sounder mode over a range of conditions including direction of transmissions, latitudes, day/night, etc. The second mode uses the same transmitter/antenna with a remote receiver located in the RF receiver package. This package is ejected directly from the pallet along the velocity vector, and its natural trajectory provides a variety of separation distances over which the experiment can be performed. Mode 1 is accomplished 3 times during the flight and mode 2, in conjunction with the long delay echo experiment, can be accomplished only once during the free-flying period when the RF receiver package is positioned properly.

The plasma flow experiment consists of a plasma generator, spherically shaped inflatable balloon, and a series of measurement instruments packaged as an integrated module. This experiment requires the use of two Orbiter provided RMSs and orientation of the Orbiter so as to place the plasma generator in the ambient plasma. The second RMS then varies the position of the diagnostic instruments so as to monitor the disturbed and ambient plasma from a variety of positions with respect to the plasma generator.

3.3 Operations Definition

3.3.1 Ground Operations

The Ground Operations include those activities required to build up a Spacelab payload into a fully integrated operational unit, as well as the post mission maintenance, refurbishment, and payload preparation for reflight. The AMPS Ground Operations flow is shown in Figure 3.3.1-1. This flow identifies the flight hardware integration site and facility locations. The ground operations activities described form the basis of the AMPS requirements for integration, maintenance and refurbishment as they relate to programmatic considerations. Facility usage, manpower, support equipment, transportation and logistics have been considered.

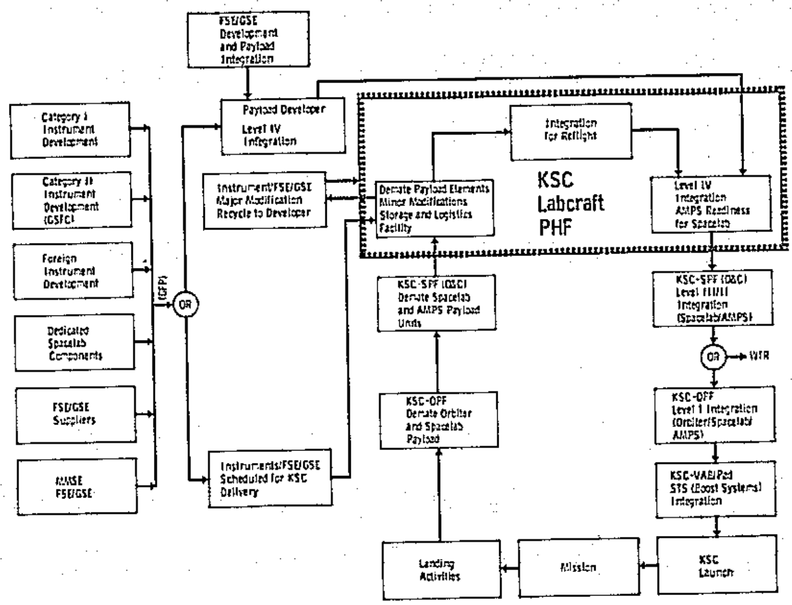


Figure 3.3.1-1 AMPS Labcraft Ground Operations

This flow is based on the requirements established in the Spacelab Payload Accommodations Handbook, May 1976; the Space Shuttle System Payload Accommodations (JSC 007700, Volume XIV, Rev D); the KSC Spacelab Operational Turnaround Allocation Schedule, April 16, 1976; and the KSC Launch Site Accommodations Handbook For STS Payloads, Rev 3, June 1976 (K-STSM-141). The integration levels shown are as follows:

(1) Level IV - AMPS payload buildup and integration test and checkout activities "off-line" from the normal Spacelab and Shuttle time critical, turnaround "on line" sequence of events.

(2) Level III - Spacelab payload buildup integrating the Spacelab pressure module, experiment racks, Payload Specialist Station (PSS) modules and AMPS pallet train onto the Automatic Checkout Equipment Stand forming the AMPS Spacelab Payload. This activity is an "on-line" Spacelab function.

(3) Level II - AMPS Spacelab payload integration test and checkout including mission sequence simulation and weight and center of gravity verification. This activity is an "on-line" Spacelab function.

(4) Level I - Integration of the AMPS Spacelab payload into the Orbiter, and the associated interface verification, which is an "on-line" activity.

After completion of these four integration levels the Orbiter must be integrated with the Shuttle Booster Systems and transported to the launch pad for launch preparation activities and final payload servicing, also an "on-line" activity.

The post mission ground operations start upon Orbiter landing. However, the first payload access will be after the Orbiter is transferred to the Orbiter Processing Facility (OPF) for removal of the AMPS Spacelab payload. Upon removal from the Orbiter the payload is transported to the Spacelab Processing Facility (SPF) for demating of the AMPS experiment racks, pallet train, and other AMPS peculiar equipment from the Spacelab pressure module, completing the on-line activities. The AMPS payload equipment is transferred back to the AMPS Payload Handling Facility (PHF) for initiation of the maintenance and refurbishment activities associated with preparation for reflight, storage or a combination of these two activities.

The following sections describe the ground operations activities and requirements for each level of payload integration. Discussions of alternate operations approaches are also included.

3.3.1.1 Level IV Integration - Payload

The primary objective of Level IV integration is to assemble the AMPS payload and perform systems level functional verification at the highest level possible to insure; 1) all payload elements, i.e. instruments, Labcraft, FSE, Payload Specialist Station modules and Spacelab experiment racks, operate satisfactorily as an integrated payload, 2) that no delay in the time critical Orbiter or Spacelab "on-line" activities will occur, and 3) a sufficient payload interface test and response data base so that payload systems health can be ascertained during the "on-line" interface checks.

The AMPS Level IV Ground Operations have been planned at the Prime Contractor's Denver Facility, for initial payload assembly and system functional verification (Figure 3.3.1-2) and at the KSC-PHF for final configuration assembly and functional verification, test, checkout and calibration of instruments (Figure 3.3.1-3) prior to delivery of the AMPS payload to the "on-line" Spacelab activities. Figure 3.3.1-4 presents a preliminary schedule for completion of these activities.

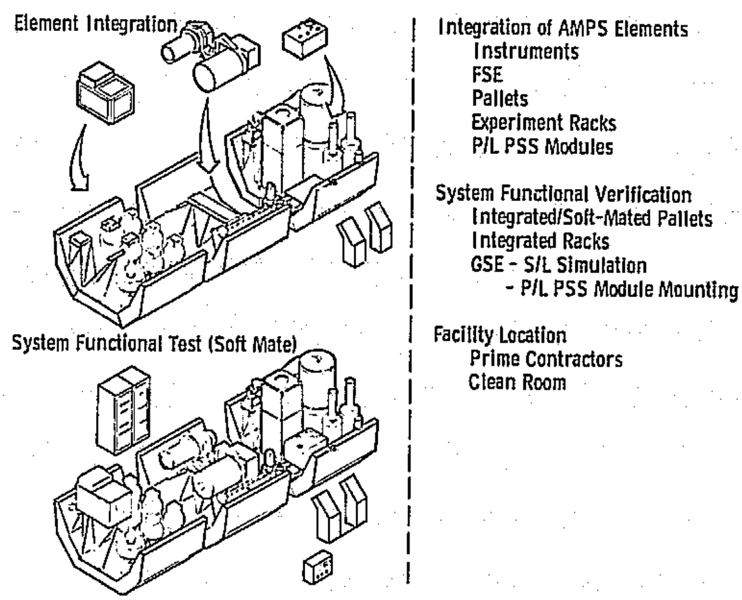


Figure 3.3.1-2 Level IV Integration - Prime Contractor's Facility

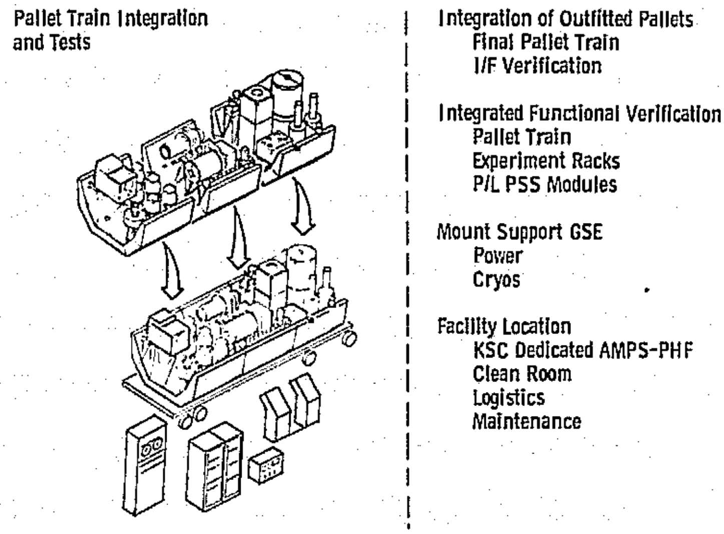


Figure 3.3.1-3 Level IV Integration - KSC AMPS Labcraft Payload Handling Facility

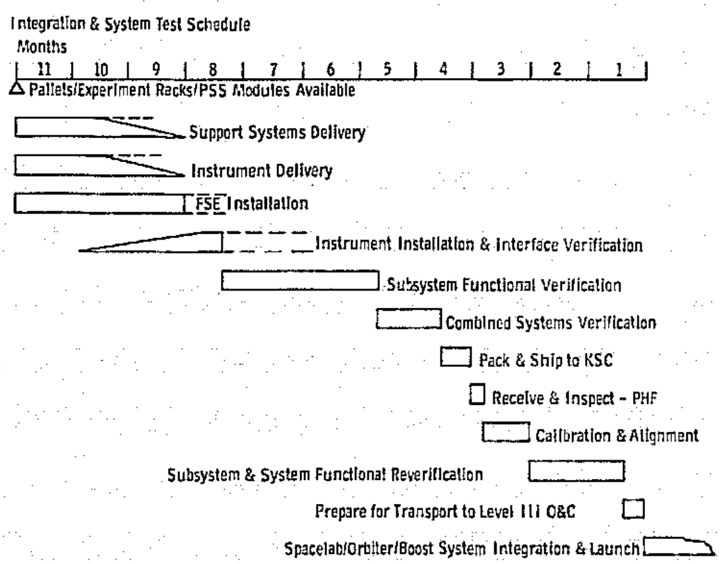


Figure 3.3.1-4 AMPS Labcraft Level IV Assembly and Checkout Schedule

Initial Payload Assembly and Verification Activities - This activity will be performed at the Martin Marietta Aerospace, Denver Division's Space Support Building in the existing High Bay Area Clean Room. This clean room meets all the space, cleanliness, and support requirements (i.e., power and cranes) necessary to assemble and test a one, two, three, four, and five pallet payload. This facility has the ability to support two or more combinations of pallet trains at one time as described in Section 5.1.2.

The initial assembly activities will start with the receipt of the GFP units which include: instruments from the instrument development contractors; instruments from a Government agency (i.e., GSFC); flight support equipment (FSE) from either a contractor or Government agency; Spacelab components (i.e., pallets, experiment racks, etc); and multi-mission support equipment (MMSE). After these items have completed receiving inspection and interface verification tests they will be readied for installation and assembly into or on the pallets with the prime contractor supplied FSE. The experiment racks and payload specialist station (PSS) module units will be assembled and each unit (i.e., individual pallets, experiment rack...) will complete system level interface verification tests. The pallets will be soft mated in a test fixture by electrically connecting the pallets together; and the PSS modules and experiment racks will be mated with and connected to the test fixture and appropriate GSE. The associated GSE will simulate the Orbiter and Spacelab interfaces necessary to perform an AMPS payload functional verification test.

The verification tests, which are described in Section 5.1.7, include the development of parametric operational data which can be used to evaluate the performance of payload equipment. These data will also be used to reverify payload compatibility and functional operation after the payload has been transported to KSC. After completion of all systems verification tests, the pallets will be demated, the experiment racks and PSS modules will be removed from the test fixture, and flight equipment will be prepared for shipment to KSC together with selected GSE.

Final Assembly and Verification Activities - This activity will be performed at the KSC-Payload Handling Facility which is yet to be identified from the various candidate facilities that already exist at KSC as discussed in Section 5.1.2. In summary, the facility requirements for the PHF include a clean room large enough to contain multiple pallet test fixtures, experiment racks, PSS module and associated GSE, and to interconnect all elements and simulate the Orbiter and Spacelab for functional verification tests.

The final assembly activities will start with receipt of the AMPS assembled pallets, experiment racks and PSS modules. After completion of the receiving inspection activities the flight elements will be assembled into the final flight configuration on the test fixture. This configuration consists of hard mating the pallet train, mechanically and electrically, and installing the experiment racks

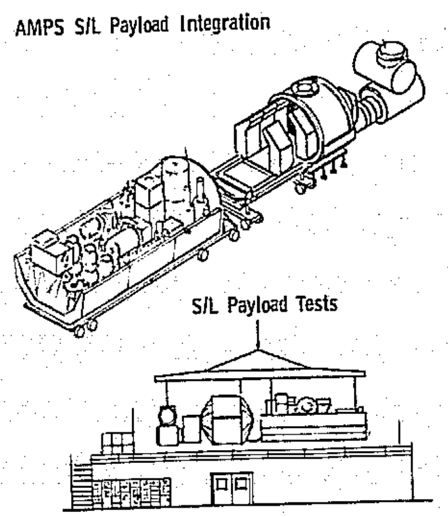
and PSS modules in the test fixture. Associated GSE will simulate the Orbiter and Spacelab interfaces necessary to conduct payload and system functional verification tests required to establish high confidence that the AMPS payload can enter the time critical "on-line" Orbiter/Spacelab integration activities. Other functions which will be performed at the PHF include: instrument checkout and alignment verification, charging of instrument cryogenic cooling systems, and mounting of special GSE on the pallet train (such as cryo charge maintenance equipment or instrument stimulus electronics).

Upon completion of this phase of the AMPS Payload "off-line" activities, the mated pallet train, experiment racks and PSS module will be transported from the AMPS-PHF to the Spacelab Processing Facility in the O&C Building at KSC for "on-line" Spacelab Level III and II integration.

3.3.1.2 Level III/II Integration - Spacelab/Payload

The primary objectives of the Spacelab "on-line" Level II and III integration activities are to: 1) assemble the payload elements into an AMPS Spacelab Payload and 2) functionally verify that the integrated Spacelab payload is operating satisfactorily and is ready to proceed with the Orbiter "on-line" Level I integration activities.

The Spacelab Level II and III integration activities are presently planned for the SPF in the KSC O&C building. All necessary support fixtures, GSE, and facility requirements to perform these integration activities will be provided and the AMPS payload can be integrated into a complete AMPS Spacelab payload using minimum AMPS unique GSE and personnel. The Spacelab Level III/II Ground Operations are shown in Figure 3.3.1-5.



Integration of AMPS and S/L Elements
- Pallet Train
- Experiment Racks
- Pressure Module
- ECT

Integrated P/L Tests
- Functional I/F Verification
- Simulated Mission Sequence

Install Safety-Critical Items
- Payload Ordnance
- Chemical Release Canisters

Facility Location
KSC, O&C, SPF

Figure 3.3.1-5 Level III/II Integration

Level III Spacelab Payload Assembly Activities - The AMPS Spacelab payload assembly will start with the receipt of the AMPS payload elements from the Level IV final integration facility, KSC-PHF. After completion of the receiving inspection activities, the AMPS Payload elements (i.e., Labcraft pallet train, experiment racks, PSS modules, and selected GSE) are installed in the Spacelab integration and checkout fixture for physical assembly of the Spacelab pressurized module elements, and AMPS payload elements into a total AMPS Spacelab payload in preparation of the Level II functional verification tests.

Level II Spacelab Payload Verification Activities - The AMPS Spacelab Payload interfaces and operations will be verified by conducting system interface verification tests, instrument interface verification, subsystem functional checkout, payload functional verification, and mission sequence simulation tests. These test and checkout activities will be performed using the assembly integration and checkout fixture and the automatic test equipment (ATE) to make up the test and integration stand.

For these two integration activities the KSC Spacelab Operational Turnaround Allocation schedule dated 16 April 1976, and summarized in Figure 3.3.1-6, identified only 46 hours of test time when electrical power would be available for payload tests. This amount of time for functional verification and checkout would severely restrict the depth and completeness of flight readiness verification which could be accomplished during these "on-line" ground operations activities. As a result of this limitation major AMPS payload operations confidence must be achieved during the "off-line" Level IV activities in the AMPS-PHF.

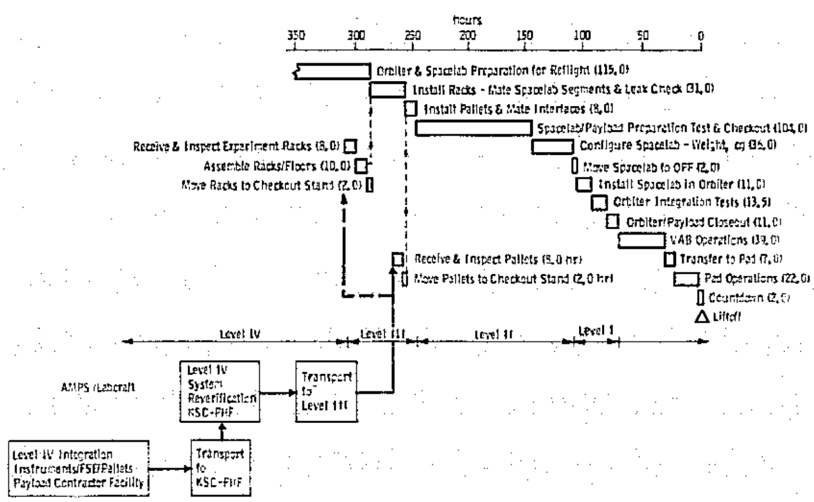


Figure 3.3.1-6 AMPS STS Ground Operations Summary Schedule

3.3.1.3 Level I Integration - Orbiter

The primary objectives of the Orbiter "on-line" Level I integration activity are to; 1) mate the AMPS Spacelab payload with the Orbiter, and 2) ready the Orbiter and payload for the succeeding launch preparations. These integration activities are accomplished in the OPF.

The AMPS Spacelab payload and Orbiter integration starts with reception of the payload and then progresses to installation in the Orbiter bay, verification of the payload interfaces, final preparation for launch and closeout of the payload bay. Upon satisfactory completion of Orbiter integration activities, the Orbiter with its AMPS Spacelab payload is transported to the vertical assembly building (VAB). The major activities are shown in Figure 3.3.1-6.

3.3.1.4 Launch Preparations and Launch

The major launch preparations include: 1) moving the Orbiter and installed payload to the VAB, 2) erecting and mating the Orbiter with the STS Booster systems, 3) towing the Shuttle flight system to the launch pad, 4) completing the final launch activities at the pad, and 5) launching the Shuttle vehicle. During these activities the payload is in the Orbiter bay with the doors closed and no payload access is permitted except after the Payload Changeout Room (PCR) is in place around the payload bay on the launch pad. During the time this PCR is in place the Shuttle Orbiter payload bay doors can be opened, if required, and access gained to the payload for minor activities requiring no power (Removal of cryogenic maintenance GSE or protective covers). This time period is approximately four hours long and occurs at approximately eight hours prior to lift-off.

3.3.1.5 Landing and Demating

The landing and demating activities are generally payload "hands-off" until the Orbiter is returned to the OPF which occurs within the first couple of hours after landing. One exception is that some items from the Spacelab, such as recorder tapes, can be removed from the Spacelab while in orbit and stowed in the Orbiter cabin, then taken from the Orbiter by the crew.

Some cirtical AMPS payload items can be removed from the Orbiter Bay in the OPF after the payload bay doors are open and the access GSE installed, but generally access to the payload should not be planned until after the AMPS Spacelab payload has been removed from the Orbiter Bay and transported to the SPF in the O&C building.

The Spacelab payload demating activities take place in the SPF starting at approximately twenty hours after landing. The Spacelab pressure module is demated and the AMPS experiment racks removed and

the pallet train is demated. The AMPS payload elements are then transported to the AMPS-PHF for maintenance and refurbishment.

3.3.1.6 Maintenance and Refurbishment

All AMPS maintenance and refurbishment activities are either initiated from or accomplished in the KSC-PHF. After receipt of the AMPS payload, instruments and FSE will be refurbished and/or updated for the next flight. The baseline plan will be to accomplish as much as possible at the PHF, but if major modifications or repairs must be made then that equipment will be returned either to a contractors' facility or GSFC for action. Preparations for the next flight will continue at the KSC-PHF for all AMPS payload elements including any newly outfitted pallets with the ground operations activities as described for the first flight. Those elements requiring storage will be stored at the PHF until their reuse is required; if however, the element will not be reused it will be sent to GSFC for permanent storage.

3.3.1.7 Alternative Approaches

Several alternative approaches to integration were studied which involved the availability of Spacelab pallets, such as: 1) pallets available for 22½ days at the contractor's Level IV integration facility, 2) pallets only available at the KSC PHF, and 3) multiple discipline payloads where another NASA center is responsible for a major element of the payload. Most of the activities described above for the AMPS payload ground operations remain the same; however, in some cases, as described in the following paragraphs, additional activities must be planned for one or both of the Level IV "off-line" integration facilities.

Pallet Availability 22½ Days (Prime Contractor's Facility) - At the request of the GSFC AMPS Project Office an alternate approach to payload assembly was evaluated. The assumption was made that the Spacelab pallets would be available at the Prime Contractor's Facility for only 22½ days prior to shipment of the assembled AMPS Pallets to KSC. Final assembly and functional verification of the payload must be accomplished in this short period. Initial assembly and verification activities, described in the baseline Level IV processes, would require the support of a pallet interface simulator as depicted in Figure 3.3.1-7. The pallet interface simulator would be used as a tool to assemble the AMPS instruments and FSE on and a test bed to perform all interface and functional verification test. After receipt of the Spacelab flight pallets the Instrument and FSE Groupings would be transferred from the pallet interface simulator to the actual flight pallets and functional interface testing completed to reverify interface compatibility. The assembled and tested AMPS payload would then follow the normal ground operations flow. The pallet simulator to flight pallet transfer schedule is shown in Figure 3.3.1-8. Analysis indicates that total impact of the transfer will be 14 days.

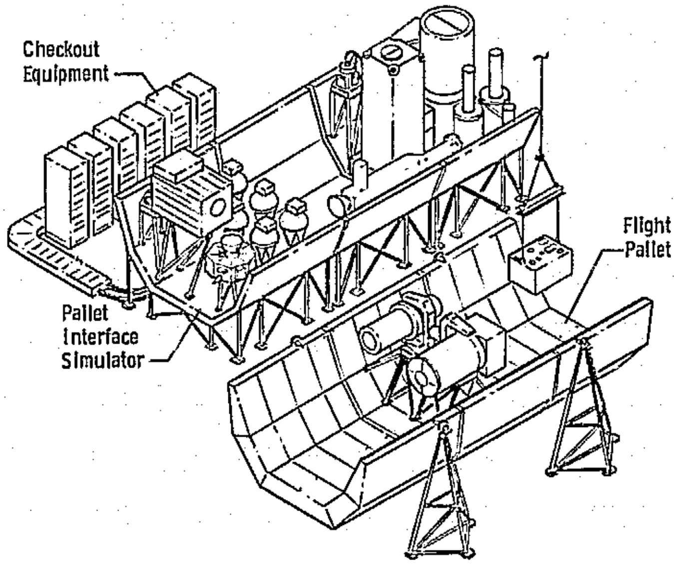


Figure 3.3.1-7 Pallet Interface Simulator - Labcraft Assembly

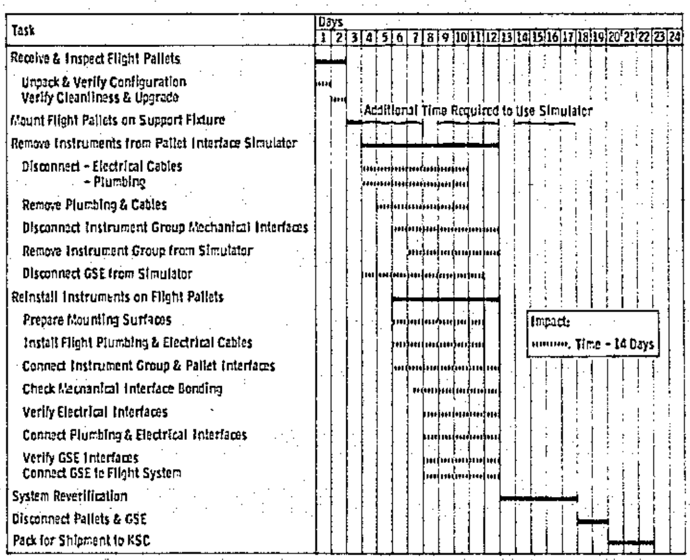


Figure 3.3.1-8 Pallet Simulator to Flight Pallet Transfer Schedule

Pallet Availability 22½ Days (KSC-PHF) - The second alternative operations approach assumed availability of the Spacelab pallets at the PHF for only 22½ days prior to transfer into the "on-line" Level III integration activities. For this case the pallet simulator would be transportable. AMPS payload instruments and FSE would be assembled on the pallet simulator and be processed through a normal initial assembly and verification. The total package (Simulator and Payload) would then transferred to the KSC-PHF for final Level IV assembly and test, which would include the transfer of instrument and FSE groups from the pallet simulator to the Spacelab flight pallets prior to completion of the Level IV activities. The schedule impact of 14 days would also apply at this facility. An alternate to equipment transfer from simulator to pallet would be provision of a pallet substructure to allow group transfer as discussed in Section 5.2.5

Multiple Discipline Payloads - The approach taken with multiple discipline payloads is essentially the same as that identified for the baseline approach. As shown in the multidiscipline payload integration schedule, Figure 3.3.1-9, the Level IV initial assembly and test would be accomplished independently for major payload segments and then transferred to the KSC PHF for Level IV final assembly and verification before entering the baseline Spacelab and Orbiter "on-line" activity sequence.

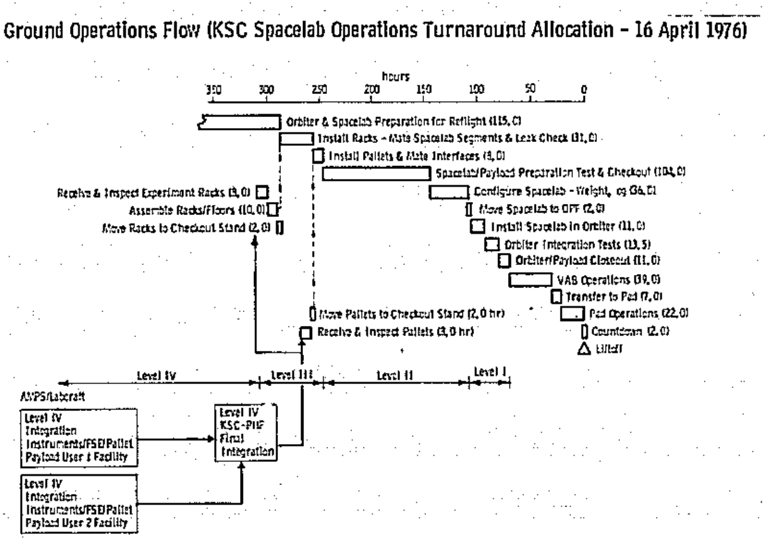


Figure 3.3.1-9 Multidiscipline Payload Integration Schedule

3.3.2 Mission Operations

The establishment of the Mission Operations concept for AMPS, which will be very similar for any Spacelab carried payload, emphasized the evaluation of four questions: (1) What are the significant elements, both spaceborne and ground that make up the operations approach? (2) What are the significant payload functions which are needed to support the operation of the mission? (3) What are the responsibilities for the participants? and (4) What are the crew participation/training requirements?

Operations Elements - Figure 3.3.2-1 depicts the elements required to operate an AMPS mission. The data generated onboard the Orbiter, within the Spacelab and by the payload is programmed to be returned via the TDRS. Commands will also be transmitted to the Orbiter, Spacelab and payload via this same system. STDN backup is a program option if required. The downlink data is divided into two groups, the first of which consists of low rate operational data (192 Kbps) from the Orbiter required to manage the overall mission from the JSC Mission Control Center (MCC). Low rate payload data, either housekeeping or science up to 64 Kbps, can be interleaved into this data stream. The second group handles instrument housekeeping and science data; up to a total of 50 Mbps digital plus video or analog up to a 4.5 MHz bandwidth. Both groups of data are received at the TDRST and routed directly to users without processing or recording. Defining the approach to data routing from the TDRST involved several considerations: required data to properly operate the mission from the MCC and the Payload Operations Control Center (POCC), expected data rates, landline data handling capability, complexity of the payload, real time and near real time science data processing and display requirements, capability to reprogram the mission based on scientific results, payload contingency replanning, location of the payload control center, and optimization of ground vs on-board experimental control.

The JSC/MCC has been assigned overall mission management including both Orbiter and Spacelab operations for STS missions. The 192 Kbps data stream is routed, via landlines, to the MCC where it is processed for real time or near real time display, control, processing and recording for post mission evaluation. This data provides subsystem status information as to the health and welfare of the STS and monitors any payload instrument parameter which affects the safety of crew and spacecraft. Mission contingency and reprogramming decisions are made at the MCC and commands will be generated and transmitted via the TDRSS to modify operations. Payload control and monitoring, because of the highly complex nature and variety of scientific instrumentation, has been considered as the responsibility of the Payload Operations Center and would be performed at the POCC. The expertise required to evaluate instrument data and reprogram experiments will be supplied by payload operations personnel trained in specific instrument operation and data analysis. Science data, together with instrument housekeeping data, is routed to the POCC from the TDRST and processed either for

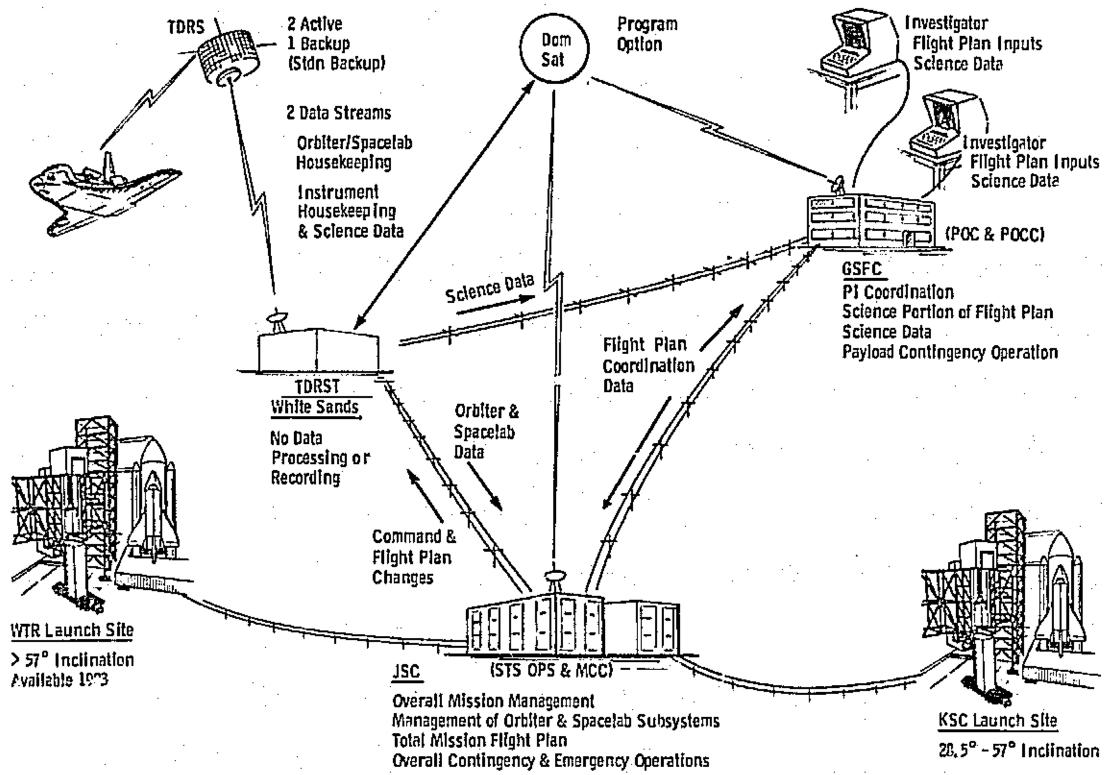


Figure 3.3.2-1 Elements of Mission Operations

real time or near real time display and/or recorded for post mission evaluation. Capability to reprogram instrument operations and generate the commands is required to modify the mission sequence. Payload scientific commands are routed through the MCC prior to transmission by the TDRS to insure crew safety is not compromised. Scientific data rates for AMPS investigations indicate a need to employ the DOMSAT, for relay transmission from TDRST to the GSFC POCC, because of a 1.3 Mbps limitation on available landlines.

The location of the POCC is based on assigned responsibility for payload definition and procurement and the need to integrate the scientist into payload operations. During the early phases of instrument and AMPS laboratory design, the development of interfaces and software for use with the Spacelab and ground checkout computers will be accommodated through communication terminals connecting to the payload operations center. These terminals can also be used to exercise end-to-end operations techniques with the POCC early in the program and with sufficient time to allow for corrections. It is envisioned, that the communication tie-in with the POCC be established as soon as possible after contracting for an instrument or flight support equipment. In addition to software and interface development,

this capability will support optimization of ground vs airborne instrument control by providing a total system simulation to exercise experiment performance and reprogramming procedures.

Operations Functions - The AMPS missions will be flown to acquire scientific data and a primary program goal is to enhance the collection and evaluation of this data. Figure 3.3.2-2 summarizes the critical functions which are necessary to provide this enhancement. The Orbiter crew and Payload Specialist functions are an integral part of the mission tasks as discussed in Section 3.2. The design of the laboratory is based on providing the capability to perform all the listed functions. The ground functions required to support overall mission performance and to control the Orbiter within the payload requirements are supplied by the MCC. The figure lists the type of tasks which are foreseen for any Spacelab payload. Examples of specific support are: Orbiter orientation planning to fill payload needs, electrical energy monitoring and resource control, mission command sequence generation/implementation, and integration of payload command sequences. These tasks provide a basic approach to the control of any mission and are tailored to fit specific program requirements identified by mission science teams.

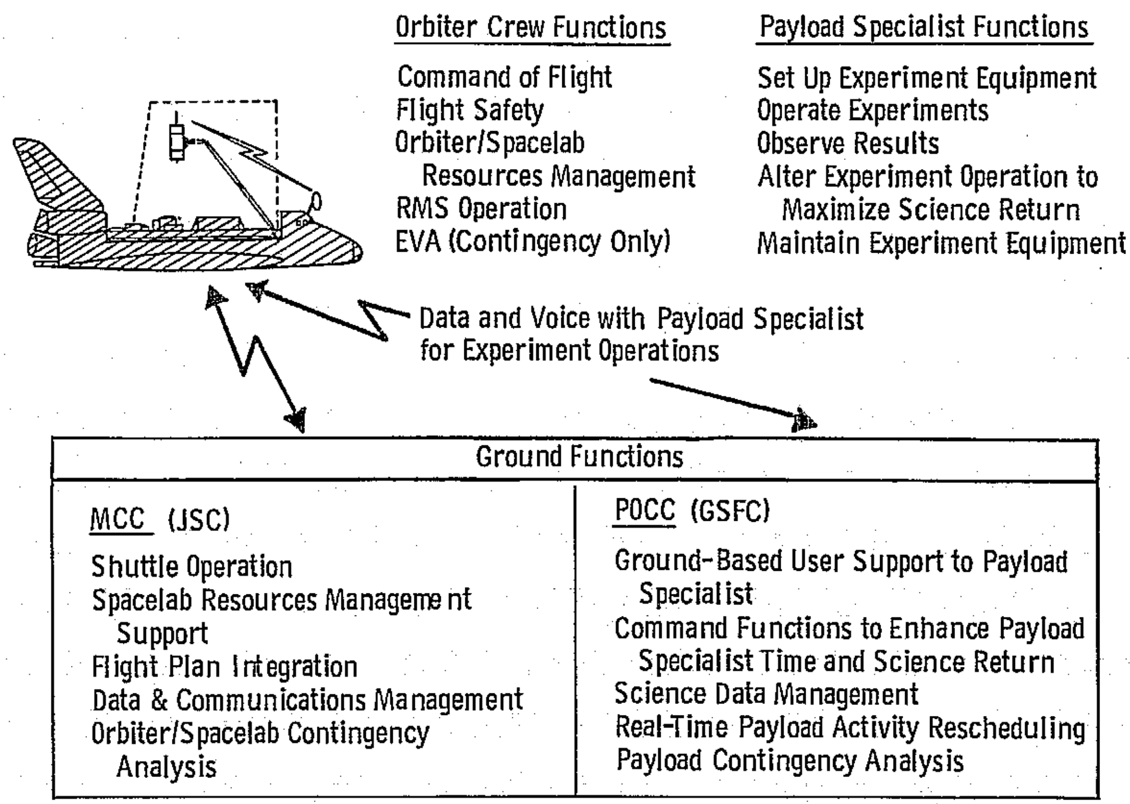


Figure 3.3.2-2 AMPS Operation Functions

The ground functions required to support experiment operations are listed in the figure and are the responsibility of the POCC. Ground based support for the payload specialist is provided in terms of real time monitoring of critical data, evaluation of experiment results and replanning of the mission to enhance science data production. Command sequence generation capability is provided to both reprogram experiment sequencing and to optimize the use of the payload specialists time to perform necessary manual functions. Science data management is provided in the form of real time data monitoring, processing and tagging of post mission evaluation data. Tie-in with principal investigators is envisioned on a real time basis so as to enhance data evaluation and reprogramming when necessary. Real-time payload activity replanning and contingency analysis will provide a team of experts and computation capability available to the payload specialist when needed to supplement his minimal replanning capability. The overall design of the laboratory and its ground support must remain sufficiently flexible to allow for optimization of ground vs spaceborne control of experiment sequences.

<u>Operations Responsibilities</u> - Responsibility for mission operations is divided into two major areas: the Orbiter at the MCC and the payload at the POCC. Figure 3.3.2-3 portrays the results of a preliminary evaluation of the operations requirements.

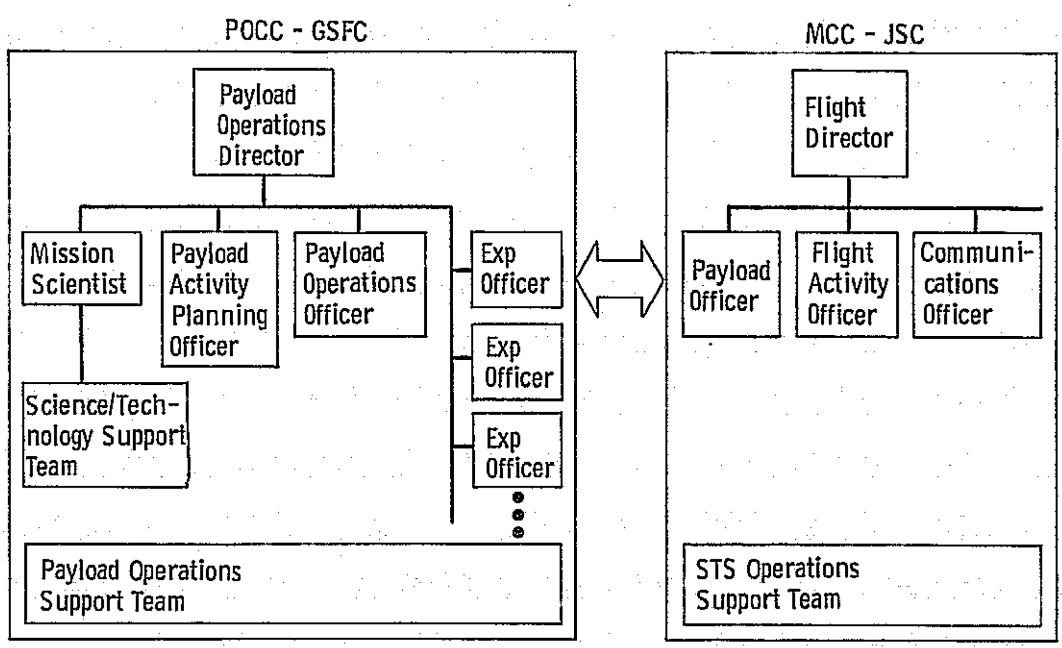


Figure 3.3.2-3 POCC/MCC Staffing

The MCC responsibilities shown address those tasks which directly impact payload operations. The Flight Director has overall responsibility for mission accomplishment and interfaces with the Payload Operations Director through his Payload officer. Flight activity planning and communications control will be integrated with payload requirements through these two MCC offices.

The payload operations officer has the responsibility for overall conduct of the scientific portion of the mission. He is supported by: the mission scientist and his staff who will be responsible for decisions affecting specific instrument usage, interexperiment priorities, and experiment replanning. They also evaluate the science data and direct mission changes to enhance the results. The Payload Activity Planning Officer is responsible for the detailed scheduling of all sequences affecting payload operations. The Payload Operations Officer is responsible for integrating both scientific and laboratory support equipment operations in terms of resource management, time allocated for a given investigation, contingency replanning, hardware usage decisions, etc. He will also be responsible for day to day interfacing with the MCC Payload Officer to resolve conflicts between mission and payload operations. The experiment officers have the responsibility to assure proper conduct of a specific investigation, and to evaluate instrument operation. They will be well versed in all phases of experiment operation including each individual instrument and will consult with the science staff in the evaluation of results and reprogramming during the mission.

Both control centers provide operations support teams for detailed analysis of subsystem and instrument performance. They will evaluate housekeeping data, flag problem areas, develop workarounds, determine maintenance approaches and generally provide technical support for the operations team.

Crew Training - The complex scientific nature of the AMPS missions, along with the limited availability of crew members, imposes a significant requirement for cross training to provide overlap for task performance. A preliminary training requirements analysis including evaluation of the types and numbers of instruments, available mission time, daily activity sequences, support equipment, operation requirements and other mission related parameters has resulted in a recommendation of a minimum training time allotment as shown in Table 3.3.2-1. This table lists, for each crewman, both the Orbiter (JSC provided) and AMPS payload related training requirements (provided by GSFC). The table suggests that each member of the crew be given selected training beyond his specific area of responsibility so that he can support other phases of the mission when required. This analysis anticipates the desirability for backup operators for each payload task but additional training may well be required in the operation of specific complex instruments, group of instruments or flight support equipment.

Table 3.3.2-1 Crew Training Time

Training Area / Crewman	Orbiter/Spacelab Operations & Subsystems (JSC Responsibility)	AMPS - Peculiar (GSFC Responsibility)	Hours
Commander/Pilot	25 Weeks at 4 hours/day (500 hours) (JSC Number)	Classroom (Exp) Classroom (FSE) Spacelab Simulator Level I Integration	28 12 12 20 72
Mission Specialist	60 Weeks at 4 hours/day (1200 hr) (JSC Number)	Classroom (Exp) Classroom (FSE) Part Task Spacelab Simulator Levels I & II Integration	28 40 20 40 40 168
Payload Specialist	8 Weeks at 4 hours/day (160 hr) (JSC Number)	Classroom (Exp) Classroom (FSE) Part Task Spacelab Simulator Levels I, II, & III Integration	28 12 20 40 80
		Total	180

*Payload specialist must be assigned to AMPS and be proficient in the science area six months prior to flight.

The training approach and related simulators identified to support AMPS scientific payload training are as follows:

(1) Classroom - Formal classroom briefings will familiarize the flight crew with overall mission objectives, basic science objectives and techniques, experiment descriptions, instrument and special flight support equipment operating techniques and simultaneous Orbiter control tasks. Control and display panel layouts and equipment operating data will be covered as a part of this training. Methodology of interfacing with ground science teams will be described.

Part Task Trainer - A simulation of the Spacelab Command and Data Management System (CDMS) will provide specific training for the operation of each specific experiment. This part task trainer supplements the Spacelab simulator,

located at JSC, which has multiple usage requirements to satisfy both Orbiter and Payload training. A Spacelab CRT and keyboard simulator along with the computational capability for active display and control, will provide actual display formats and operational sequence control.

(2) Spacelab Simulator - Integrated crew training will be accomplished on the Spacelab simulator. This simulator provides a high fidelity operational simulation for Spacelab subsystem tasks. The simulator computer system will be capable of taking AMPS payload simulation mathematical models of instruments, simulated science data outputs, and target phenomenon; and will present these data for both nominal and malfunction crew activities. This Spacelab simulator will be joined with the Shuttle Mission Simulator, Mission Control Center, and Payload Operations Control Center for integrated mission simulations prior to final pre-flight simulations using the AMPS flight hardware at KSC.

(3) Test/Integration - Of particular importance to the training of the crew will be familiarization with the equipment which they must operate and control during the mission. A "hands-on" attitude early in the development phase should be instituted as a program requirement. The payload specialists should be assigned to the program during the initial design phase and prior to the point of instrument verification. He should continue through level IV integration and checkout to become familiar with the characteristics of each instrument and piece of flight support equipment he must operate during the mission. He should also be involved in level III, II and I integration activities to evaluate the effect of interfacing with the Orbiter. The mission specialist should be assigned prior to level IV integration and checkout so that he can become familiar with his science related support tasks. He will provide consultation regarding operation of STS selected hardware. The level III, II, and I integration and checkout phases will also provide for the integration of the selected flight crew members as to the conduct of the mission and the opportunity to refine their expected mission activities.

3.4 Experiment/Instrument Definition

To serve as a basis for payload definition and evolution, a sequence of strawman scientific missions was developed by the AMPS Science Working Group. These missions consist of a set of scientific objectives for five separate flights, a group of experiments which address these scientific investigations for each flight, and a complement of instruments which are required to implement the indicated experiments. The mission objectives and corresponding experiment groups are summarized in Table 3.4-1 for all five flights. On the basis of these starting data, several analyses were conducted. First, a study of each experiment was carried out and an implementation plan developed. These plans allow the development of the operational requirements for each flight to be accomplished, and are described below. Second, a detailed analysis of the compatibility of the instrument/support equipment hardware/operational requirements with the Orbiter and Spacelab capabilities was performed for these five missions. On the basis of these studies, optimum payload configurations were developed, along with a complete set of ground and mission operational profiles. The results of these studies are reported in later sections of this volume. In the remaining discussions, each flight is described, including the mission objectives, the experiment/instrument complement, the experiment implementation plans, and the resulting payload configurations and integrated mission profiles.

Table 3.4-1 Mission Scientific Objectives/Experiment Summary

Objectives \ Flight No.	1	2	3	4	5
Study of Strato-spheric/Mesospheric Composition, Energy Budget, and Chemical/Transport Processes	Minor Constituent Mapping	Minor Constituent Mapping	Transport Effects on Minor Constituent Distributions	Transport Effects on Minor Constituent Distributions	Determination of Fundamental Atomic/Molecular Rate Parameters
Investigations of Magnetospheric/Ionospheric Coupling Processes	Generation of Artificial Gravity Waves	Ionospheric Conductivity Modification HF Wave/Particle Interactions	$\underline{E} \parallel \underline{B}$ Origin Ionospheric Conductivity Modification	$\underline{E}$ and $\underline{B}$ Field Mapping HF Wave/Particle Interactions	Generation of Field-Aligned Currents VLF Wave/Particle Interactions
Study of Fundamental Plasma Processes	Beam-Plasma Interactions	HF Wave Injection/Instability Generation	Beam Excitation of Atmospheric Emission	Beam/Plasma Wave Generation	Beam Excitation of Auroral Processes Alfven Wave Generation
Determination of Physical Processes Associated With Plasma Flows	Shuttle Wake Studies	Test Body/Plasma Flow		Test Body/Plasma Flow	V-Critical Study

<u>Experiment Descriptions and Implementation Plans</u> - In order to properly plan and effect the desired scientific/experiment goals of the strawman missions, a number of operational and Orbiter/Spacelab support requirements must be established. The requirements for

Orbiter/Spacelab attitude and pointing for desired target locations; the estimates of data samples, integration times and repetition rates; the deployment requirements including the needed ejection ΔV and desired range; and the need for correlative supporting data acquired from ground-based or other spaceborne platforms must all be considered and established. In addition, the orbital scenario required, including altitude, inclination, launch data and times, as well as lighting conditions and other timing requirements, all provide the background setting within which a set -by-step conduct of each experiment can be established. In order to derive these operational requirements, individual experiment implementation plans have been developed for each experiment required by each mission. These plans graphically sketch the manner in which the experiment proceeds.

For certain experiments, instrument functional timelines are adequate to derive the time-dependent Orbiter/Spacelab support requirements. In other experiments, details of orbital mechanics, lighting conditions required, and deployment/maneuvering needs are described. With the experiment implementation plans for each experiment, the mission requirements for a completely integrated, compatible group of experiments may then be established, including crew participation. Further studies involving payload design and optimization may then proceed.

3.4.1 Flight 1 Experiment Plan

For flight number one, the mission-level objectives set forth by the AMPS Science Working Group consist of:

- o study the source mechanism, propagation characteristics and other properties of naturally occurring gravity waves in the Earth's atmosphere by generating an artificial gravity wave;
- o study the nature of energetic particle beam interactions with the neutral and ionized atmosphere in the vicinity of the Shuttle; and
- o study the role of trace constituents in the chemistry and dynamics of the atmosphere by remote excitation and sensing and provide diagnostic measurements for plasma/atmosphere interaction experiments.

The experiments designated by the AMPS Science Working Group which address these and later objectives include: (a) Acoustic Gravity Wave Generation, (b) Atmospheric Minor Constituent Profiles, (c) Electron Beam Studies, (d) EMI and Orbiter Wake Mapping, (e) Particulate and Gas Effluence Contaminant Studies, and (f) Solar Flux Monitor Calibration.

The corresponding summary of experiments and required instruments is shown in Table 3.4.1-1. On the basis of these data, and data

acquired from the Experiment Operations Requirements (EOR) document generated by the AMPS Science Working Group, the following experiment descriptions and implementation plans were derived to serve as a basis for deriving systems, subsystems and mission operations requirements.

Table 3.4.1-1 Flight 1 Experiment/Instrument Summary

Experiment	IFRD No.	Short Title
Acoustic Gravity Wave	I-21 II-3	Gas Release Module OBIPS (L^3 TV)
Electron Beam Studies	I-9 III-3 II-3 III-2 III-4	Electron Accelerator Gas Plume Release System OBIPS (L^3 TV on RMS) Vector Flux Gate Magnetometer Faraday Cup, Electrostatic Analyzer, Plasma Potential Probe
Atmospheric Minor Constituents Profiles	I-1 II-7 II-9 II-10	Laser Sounder Cryo-Cooled Far-IR Radiometer Near-IR Spectrometer Cryo-Cooled Interferometer/ Spectrometer
EMI Field Mapping and Space Shuttle Wake Measurements	III-25 III-18 III-23	E-Field Power Spectral Density Analyzer B-Field Antenna and Receiver Langmuir Probe Ion Mass Spectrometer Vector Magnetometer Planar RPA Neutral Mass Spectrometer
Particulate and Gas Effluence	NA	Integrated Environmental Contamination Monitor
Solar Flux Calibration	IV-1	Array of Spectrophotometers (300 to 3500 A)

Acoustic Gravity Wave Generation - Acoustic gravity waves (consisting of low-frequency bulk wave motions in the upper atmosphere that propagate under the restoring force of the Earth's gravity field) provide an important energy transfer process of atmospheric dynamics. This AMPS experiment is designed to study the generation and propagation mechanisms of these wave fields. The plan calls for the high-altitude (200 $\pm$ 50 km) release of a significant amount of a neutral gas (70 kg of nitrogen) injected at orbital velocity near sunset.

The gas is released by a module ejected from the payload with a ΔV of +5 m/sec and deployed at a safe separation distance from the Orbiter. The release expansion takes less than a second and provides a virtual point source for the rapidly expanding gas cloud as it interacts with the ambient atmosphere and is braked to a halt. The momentum/energy pulse generates a propagative wave field observed by the backscatter radar of the Arecibo Ionospheric Observatory and by a three-station network of RF sounders. The experiment, performed six times on the first flight, requires six gas release modules. A repeating ground track requirement sets the orbital altitude at 209 km. The magnitude and altitude of the releases are to be held constant, while the horizontal range of the release points to the ground radar is varied (2 at 0 km, 2 at 100 km range, 2 at 200 km range).

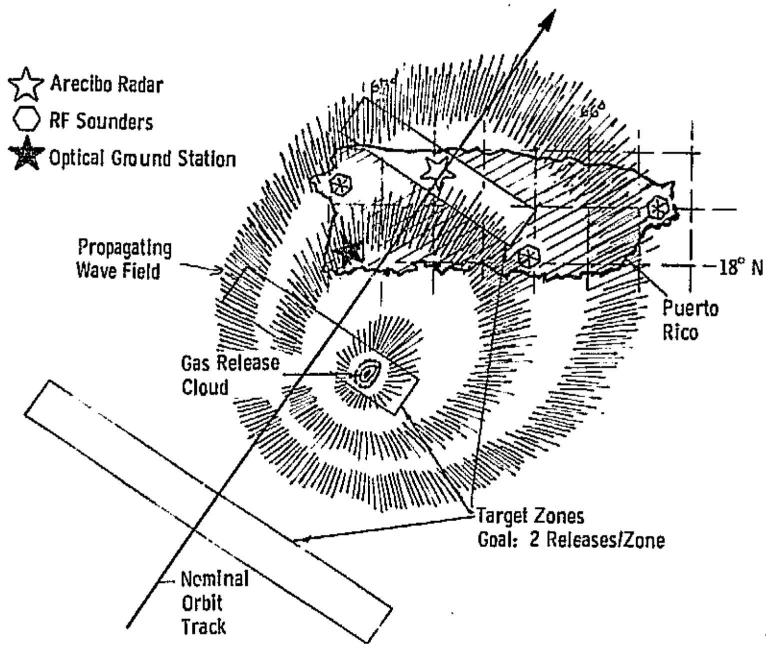


Figure 3.4.1-1 Gas Release Target Zones

The initial phase of the gas cloud expansion is optically observed both from a ground station and from the AMPS payload to study the dynamics of the release and initial interaction with the atmosphere. The release is seeded with an optically active material to make the release visible by solar resonant scattering to both stations, which are equipped with low-light-level (L^3) TV cameras. Subsequent observations employ the radar and RF sounders to map the propagation and decay of the traveling wave field.

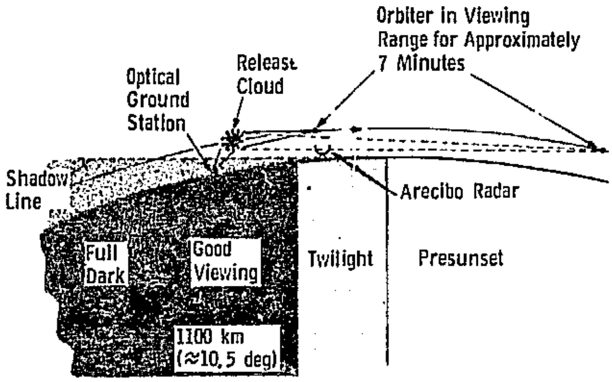


Figure 3.4.1-2 Conditions For Optical Viewing

Electron Beam Studies - Energetic charged particle beams are an important tool for many areas of AMPS magnetospheric and plasma physics experimentation. A primary objective of Flight 1 is to determine the operating characteristics of the electron beam from the Orbiter and to study spacecraft charging and neutralization. This investigation is performed using two modes, both of which utilize the electron accelerator. In mode 1, just prior to firing the electron beam along the +Z axis of the Orbiter, a plume of nitrogen gas is released by a pallet-mounted gas release system as indicated, under dark conditions.

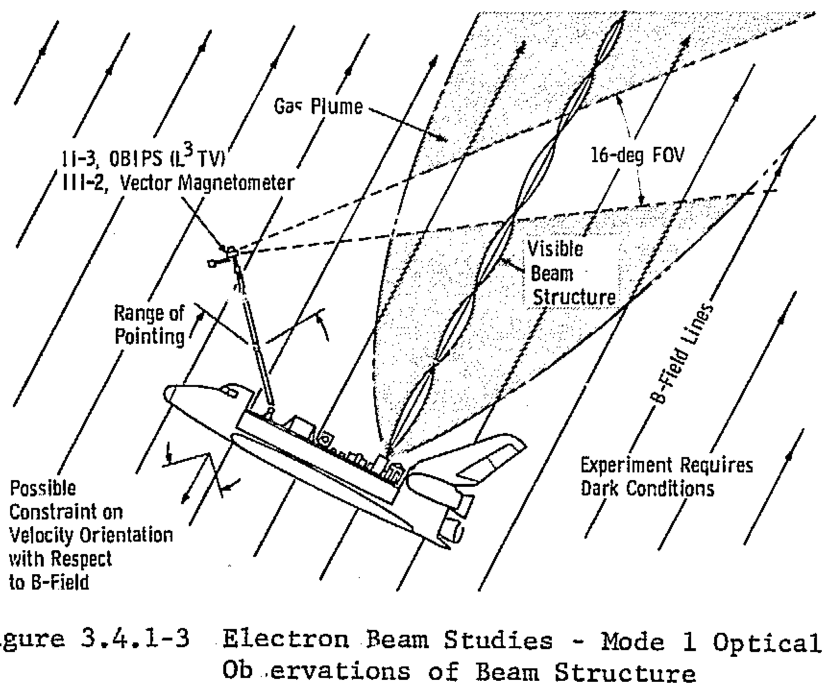


Figure 3.4.1-3 Electron Beam Studies - Mode 1 Optical Observations of Beam Structure

As the plume disperses, the electron beam is fired through it. Interactions of the energetic electrons with the nitrogen molecules give a visible indication of the beam structure and characteristics against a dark sky background. The beam is observed by the onboard directable low-light-level TV camera mounted on the end of the RMS. Relative pitch angles of the beam with respect to the geomagnetic field vector are monitored by a vector magnetometer also mounted on the RMS to reduce the effects from Orbiter-generated magnetic fields. The TV camera is also used during this mode to observe the Orbiter for any visible corona-type discharges of large electric fields caused by charge buildup. These indications are correlated with vehicle potential measurements made by the mode 2 beam diagnostic package, which includes a Faraday cup to measure beam current, an electrostatic analyzer to measure particle energies, and a plasma potential probe. The beam direction with respect to the magnetic field, as in mode 1, is monitored and controlled by the fluxgate vector magnetometer. Mode 2 relies on monitoring instruments inserted into the beam and does not require dark conditions. It should be noted that all of the diagnostic instruments for both modes 1 and 2 are combined into a single instrument module mounted on a 2.5-meter extension boom for deployment by the RMS.

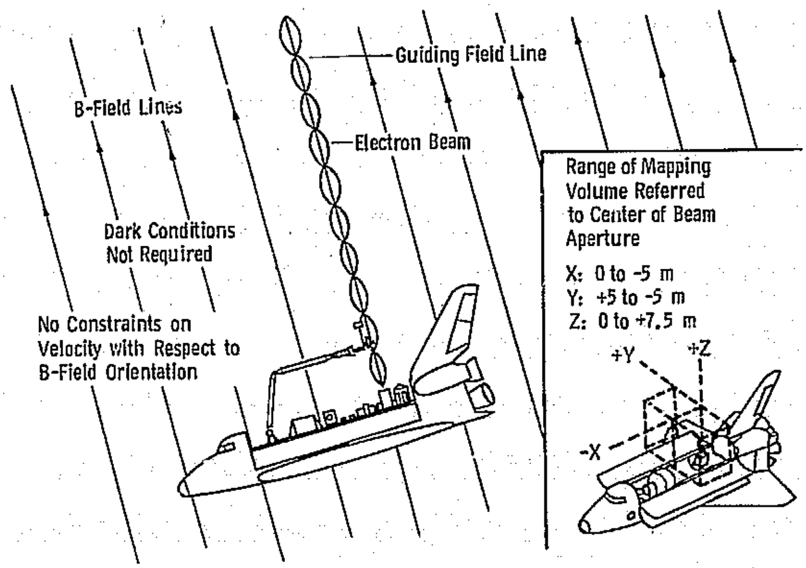


Figure 3.4.1-4 Electron Beam Studies - Mode 2 Mapping by In Situ Diagnostic Sensors

Atmospheric Minor Constituent Mapping - A long-term objective of the AMPS program within the atmospheric sciences is concerned with the understanding of the role of key minor constituent abundances and

distributions, and the possible impact of their variations on both terrestrial life and on the general climate. In particular, the consequences of man-made catalytic agents introduced within the troposphere on significant reductions of stratospheric ozone, and the effect of such reductions on increased UV radiation with its potential effects on terrestrial life, is a matter of urgent concern. In addition to the potentially deleterious effects on mankind, the consequences of significant reduction of stratospheric ozone on climate appear to be significant, and requires careful study. An understanding of these phenomena requires a carefully programmed series of investigations. A broad goal, assigned to AMPS in this area is the determination of stratospheric/mesospheric minor constituent altitude profiles, and the study of the dynamics of the mesosphere/lower thermosphere. However, a more specific goal is desirable. For this purpose an observational test of photochemical/transport models by obtaining a high spatial resolution global ozone map has been selected as a problem for the assessment of the mission operational requirements.

The key elements to be determined in the study of this problem are identified in Figure 3.4.1-5. Many parameters contribute to the observed global distribution of minor constituents. The incoming solar ultraviolet radiation drives a wide variety of photochemical reactions, producing a number of relevant minor species. The solar UV radiation also produces atmospheric heating, contributing to the observed atmospheric temperature structure. The thermal structure in turn drives the motions which are inherent to the dynamical behavior of these atmospheric regions, and also controls the rate of many photochemical and chemical reactions through the temperature dependence of the rate coefficients. Finally, both natural and man-produced catalytic agents introduced into the troposphere are transported to the stratosphere/mesosphere where they impact the resulting chemical processes and abundances. All these factors combined according to their relevant spatial and temporal scales to produce the global distributions of minor constituents.

Considerations of estimated concentrations, instrument signal-to-noise ratios, estimates of sampling requirements, and considerations of various transport scales lead to the need for a 5 degree by 5 degree mapping grid for the distribution of the minor constituents, such as ozone. In order to obtain statistically reliable samples of data, essentially full time operation of the indicated instruments is required for each seven-day flight, as shown in Figure 3.4.1-6. A hard-mounted, nadir-pointed laser sounder is used to determine OH, O_3 and NO profiles by means of two tunable, UV dye lasers. A cryogenically cooled limb scanning radiometer maps the vertical profiles of H_2O, CH_4, N_2O, HNO_3, NO_2, ClO, and the chloroflouromethanes. A co-aligned cryogenically cooled interferometer/spectrometer obtains data for analysis of suspected constituents with abundances in the parts per billion and trillion range.

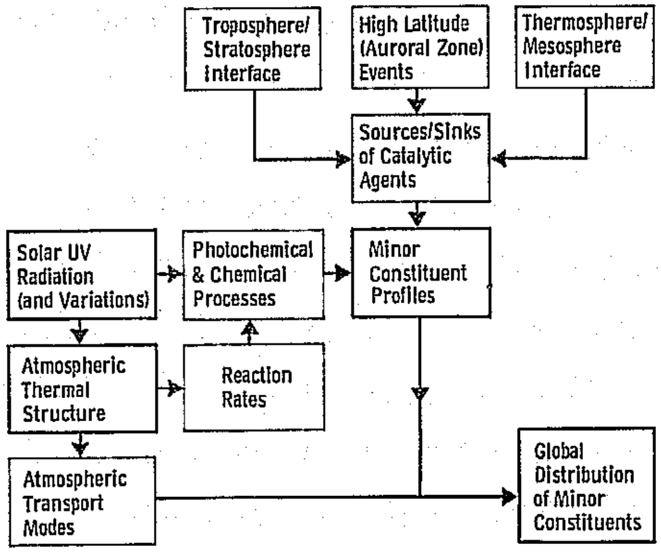


Figure 3.4.1-5 Elements of Minor Constituent Investigation

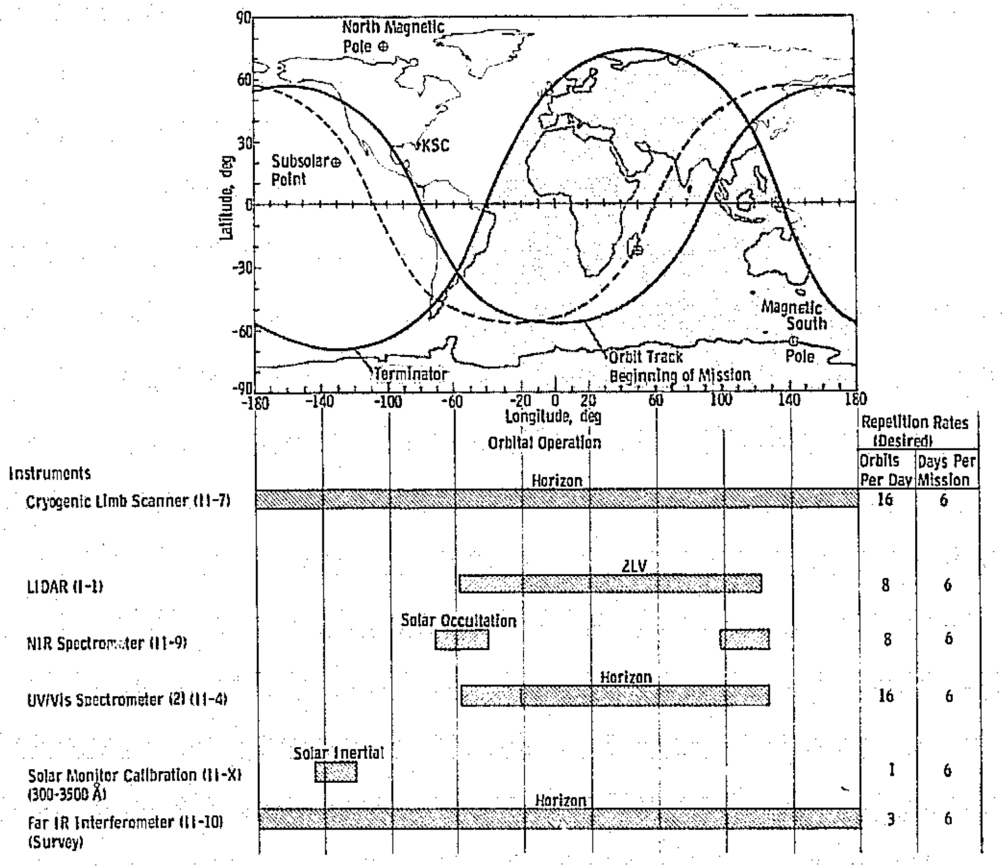


Figure 3.4.1-6 Minor Constituents Integrated Instrument Timeline

Finally, a near infrared spectrometer used in a solar occultation mode obtains vertical profiles of HCl, OH, and N_2O, for this sample problem analysis. As indicated in the orbital instrument timeline, the entire array of instruments operates during all available orbital time, and in unison from pointing platforms which permit simultaneous viewing of multiple target areas in order to provide the world-wide grid of data. On the basis of these data, the following results are expected: (1) identification of dominant transport modes in the lower stratosphere, (2) a determination of the fluxes of Nitric Oxide transported from the high latitude auroral regions to mid-latitudes, (3) identification of fine structure (secondary maxima) in the ozone profiles, (4) a determination of chemical/photochemical relaxation times from diurnal variations of abundances, (5) a determination of vertical fluxes of N_2O from the troposphere into the stratosphere, and (6) a measure of the vertical and latitude profiles of eddy transport coefficients.

Electromagnetic Interference (EMI) Mapping - EMI field mapping determines the electromagnetic noise environment of the combined Orbiter and AMPS Spacelab payload. This information is essential for planning high-sensitivity experiments in later flights. The close-in space about the payload bay is mapped by a diagnostic EMI package deployed and traversed on the RMS as shown in Figure 3.4.1-7. This diagnostic package includes a broadband single-axis E-field dipole, a single-axis B-field monitor, a Langmuir probe, and an ion mass spectrometer. After the close-in mapping is completed, this module is ejected from the payload as an environmental sensing probe to obtain a profile of the EMI field at greater distances.

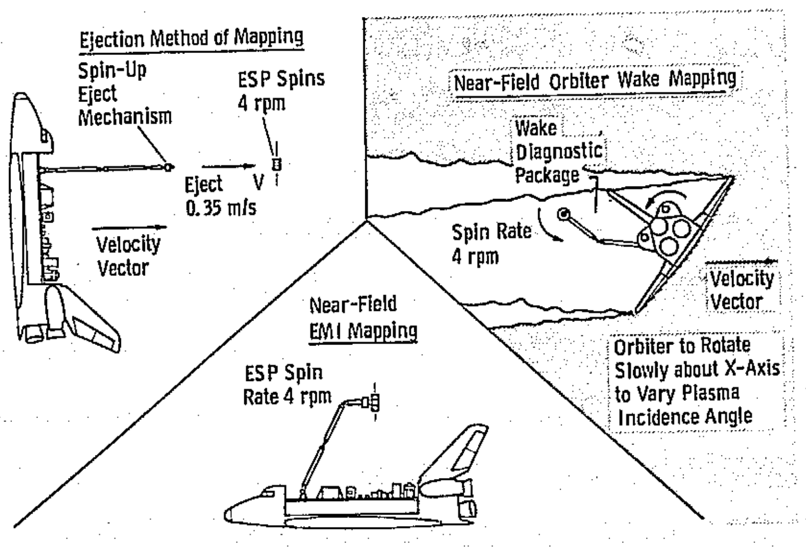


Figure 3.4.1-7 EMI and Orbiter Wake Mapping

Orbiter Wake Mapping - Understanding the plasma wake characteristics of the Orbiter provides a foundation for plasma wake experiments scheduled for later flights. The complete EMI mapping instruments serve here in combination with a planar RPA, and a neutral mass spectrometer. As illustrated here, the Orbiter attitude with respect to the ambient plasma flow is the primary variable for both close-in and far-field mapping, as the Orbiter slowly rotates about its X-axis. Upon completion of near-field measurements, the ejection trajectory traverses the Orbiter downstream wake several times at several kilometers range.

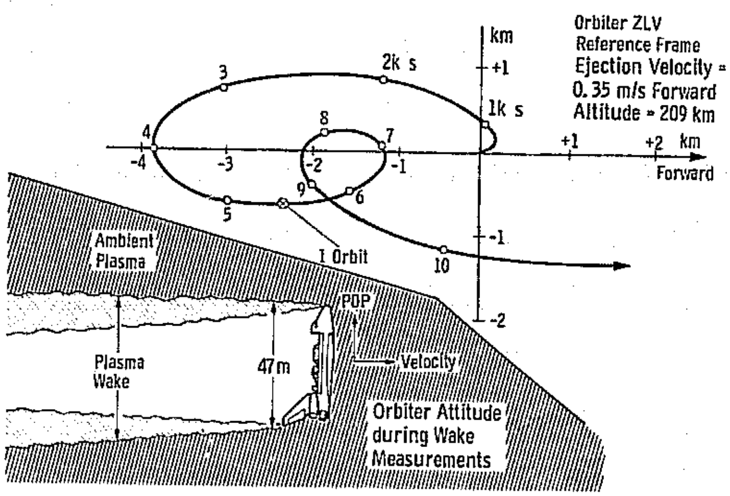


Figure 3.4.1-8 Flight 1 - Orbiter Wake and EMI/Far-Field Mapping Trajectory

Particulate and Gas Effluence Mapping - The integrated environmental contamination monitor developed for Orbiter flight tests is used to define the baseline environmental conditions required to support experiment planning in later flights. This package contains a variety of instruments including a mass spectrometer, photometers to measure particle-scattered light and ambient gas and particulate cloud column densities, and quartz crystal microbalances to measure the surface contamination buildup. The mass spectrometer mounted with the EMI package provides additional in situ information on neutral gas species at locations outside the payload bay.

Absolute Solar Flux Monitor Calibration - In support of the analysis of long-term calibration capabilities of solar flux instruments deployed on automated satellites, an array of spectrometers, covering the spectral range from 300 to 3000 angstroms (Figure 3.4.1-9), is included as part of the AMPS instrument payload on AMPS flight one, and all

subsequent flights. These instruments are bore-sighted with a sun tracker and mounted on a limited-range gimbal that permits the Orbiter to be put into a free drift mode while the solar measurements are made, at the desired rate of once per day. As indicated in the minor constituents discussion, the solar ultraviolet flux and possible variations, either differential or absolute, are important to the photochemistry and thermal structure of the stratosphere/mesosphere. In order to study possible correlations of relevant atmospheric parameters with possible solar flux variations, the calibration monitor is operated in coordination with the other required instruments as indicated in the orbital timeline in Figure 3.4.1-6.

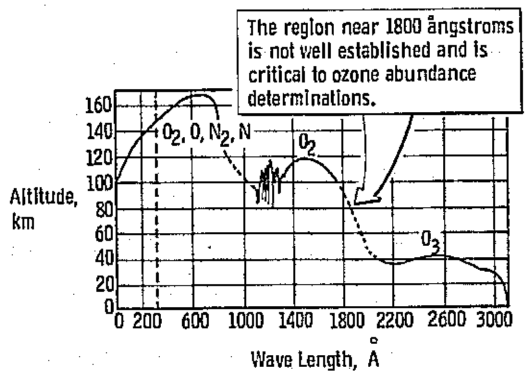


Figure 3.4.1-9 Solar Ultraviolet Flux Incident on Stratosphere/Mesosphere

3.4.2 Flight 2 Experiment Plan

The mission-level scientific objectives for Flight 2 which have been developed by the AMPS Science Working Group include:

- o the investigation of the circuit and generator characteristics of the ionosphere-magnetosphere system by perturbing the natural conductivity with a gas release;
- o the study of physical processes in the ionospheric plasma by injection of electromagnetic waves;
- o the study of the fundamental physical processes associated with a supersonic/hypersonic motion of geometrically simple bodies through a rarefied plasma; and
- o the study of the role of trace constituents in the chemistry and dynamics of the atmosphere by remote excitation and sensing.

The experiments designated to address these objectives are: (a) conductivity modification, (b) wave/particle interaction, (c) long-

delay echo, (d) plasma flow, and a continuation of, (e) atmospheric minor constituent profiles, and (f) absolute solar flux calibration experiments. The corresponding list of experiments and required instruments is shown in Table 3.4.2-1. With these data and EOR data, the following experiment descriptions and implementation plans were derived for this second flight. Again these plans serve as the basis for deriving the systems, subsystems and mission operational requirements.

Table 3.4.2-1 Flight 2 Experiment/Instrument Summary

Experiment	Instruments	
	IFRD No.	Short Title
Conductivity Modification	I-21 II-3 II-4	Checmical Release Module OBIPS (L^3 TV) UV Spectrometers
Wave Particle Interactions	I-12 III-2	RF Plasma Wave Package Flux Gate Vector Magnetometer
Long-Delay Echo	I-12 III-2 I-12 III-2	RF Plasma Wave Package Flux Gate Vector Magnetometer Subsatellite RF Plasma Wave Receiver Subsatellite Flux Gate Vector Magnetometer
Plasma Flow	III-17 III-2 III-10 III-18 III-22 III-23	Deployable Test Body (Spere) Flux Gate Vector Magnetometer Ion Mass Distribution Analyzer Planar RPA Langmuir-Type Current Collector Neutral Mass Spectrometer
Atmospheric Minor Constituents Profiles	I-1 II-7 II-9 II-10 II-4	Laser Sounder Cryo-Cooled Far-IR Radiometer Near-IR Spectrometer Cryo-Cooled Interferometer/ Spectrometer UV-Visible-Near-IR Spectrometer/ Photometer
Solar Flux Calibration	IV-1	Array of Spectrophotometers (300 to 3500 Å)

<u>Chemical Release for Conductivity Modification</u> - A better understanding of the ionosphere electrojet-current systems associated with the auroral zones provides an insight to the flow of distant magnetospheric currents generated by the interactions of the solar wind with

the magnetosphere, as well as the effects of these intense currents on the upper atmosphere. The plan for this experiment is to produce a region of increased conductivity in the auroral zone at an altitude of 150 to 200 kilometers by the pyrotechnic release of a barium-copper-oxygen mixture at a uniform rate of 10 kg/km along a 100-kilometer trail. This material expands to a cylindrical cloud on the order of 12 to 20 kilometers in diameter and over 100 kilometers long. The chemical is released at the highest northern latitude part of the orbit in the vicinity of Fort Churchill, Manitoba, as shown in Figure 3.4.2-1. The long dimension of the cylindrical cloud is oriented east-west and parallel to the auroral zone arc. The chemical is released just after sunset so the initial phase of the release can be observed from ground optical stations located at Fort Churchill and at Port Nelson approximately 200 kilometers SSE. The barium is ionized by the sunlight to produce the required conductivity modification in the release zone. Coordinated ground observations conducted at Fort Churchill throughout the night to observe effects of the release include visual observations of auroral phenomena, use of radio frequency sounders, fluctuations in the electric and magnetic fields, and in situ sounding rocket probes for measurements of E fields, currents, and precipitating particles. The next orbit pass (approximately 90 minutes later) 200 to 400 kilometers to the south of the release zone affords an excellent viewing geometry. The onboard low-light-level TV and the two Ebert Fastie spectrometers are used to diagnose the auroral processes in the region of the release as indicated. Both instruments are on pointing platforms and are directed at features of interest by the payload specialist.

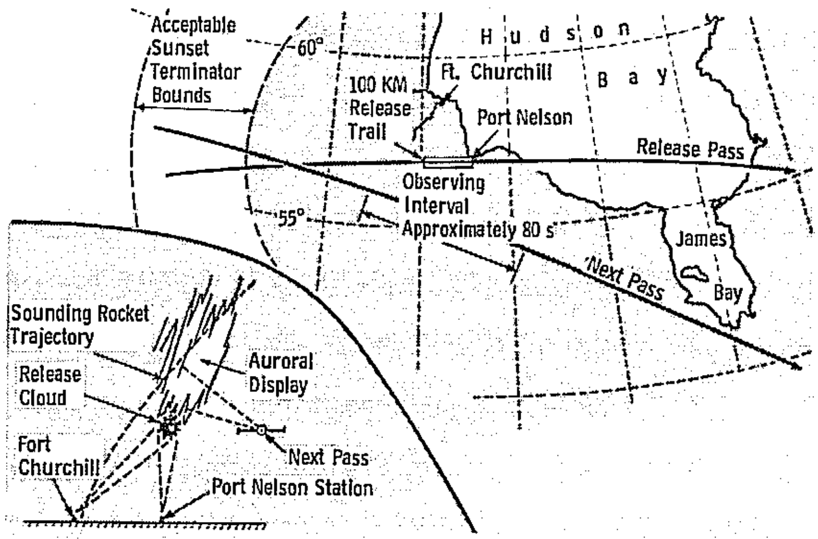


Figure 3.4.2-1 Location of Conductivity Modification Release

<u>Wave/Particle Interactions</u> - An experiment goal is to study the interactions between electromagnetic waves and the ionospheric charged

particles over the frequency ranges where resonant coupling is expected to occur (30 KHz to 20 MHz). Implementation of the experiment incorporates two modes. The first mode uses a single-ended sounder technique and the second mode uses a receiver located on a subsatellite.

Mode 1 requires a transmitter and receiver coupled with a 100-meter tip-to-tip dipole antenna. Orientation of the antenna with respect to the geomagnetic field is monitored by a fluxgate vector magnetometer deployed on the RMS. The instrument is operated in a sounder mode over a wide range of ambient conditions including orientation with respect to the geomagnetic field vector, high and middle latitudes and day/night variations.

Mode 2 requires the same transmitter and dipole antenna on the Orbiter and also requires a receiver and a 10-meter tip-to-tip dipole antenna on the subsatellite. Attitude of the antenna with respect to the geomagnetic field vector is again monitored by the fluxgate vector magnetometer. During this mode of operation, propagation-type measurements are taken using the Orbiter-mounted instrument as the source with the receiver on the subsatellite. After the subsatellite is deployed from the Orbiter, the total observation time available is on the order of one to two orbit periods as shown in Figure 3.4.2-2. This operating period will also be shared with the long-delay echo experiment described below.

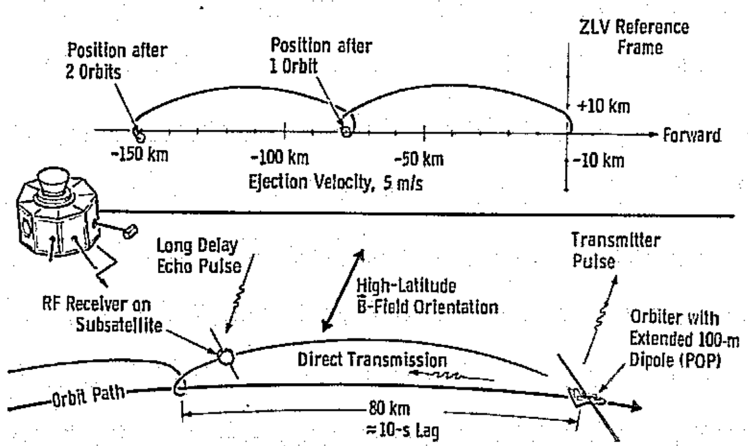


Figure 3.4.2-2 HF Transmitter/Long-Delay Echo and Wave/Particle Interactions

<u>Long-Delay Echo</u> - In this experiment signals are transmitted in the nadir direction, and echo returns with a time delay on the order of 10 seconds are observed. The free-flyer subsatellite instrument complement is the same as for mode 2 of the wave/particle interaction measurements. The orbital dynamics of the deployment allow the free flyer to pass through the region approximately 80 kilometers behind the Orbiter at the optimum location for the experiment.

<u>Plasma Flow</u> - With the data and understanding of the Orbiter wake from Flight 1 in hand, this experiment studies the characteristics of the wake of an axially symmetric test body (4-meter radius sphere) with a conductive surface and maintained at a known, controllable potential with respect to the ambient plasma. The wake generator body consists of an inflatable metalized balloon mounted on the end of a 5-meter extension boom that, in turn, is deployed on the end of one RMS.

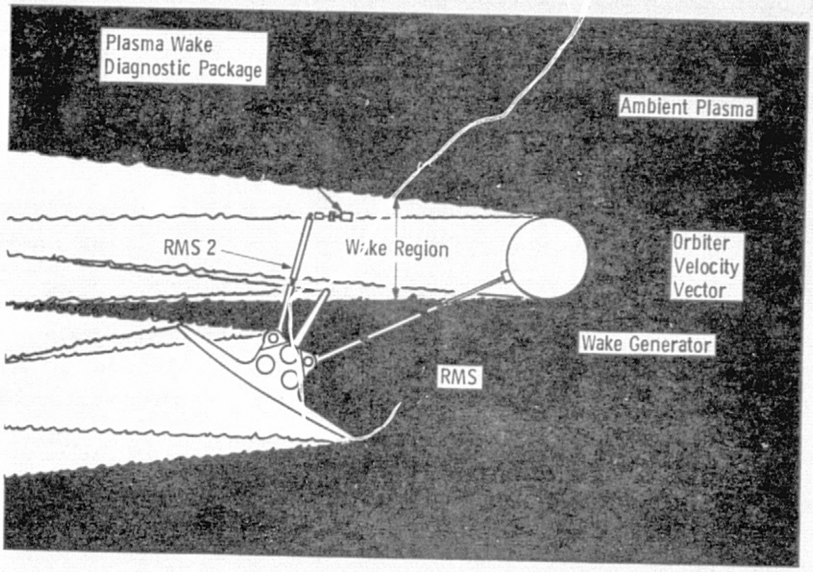


Figure 3.4.2-3 Plasma Flow Experiment Configuration

The vehicle is oriented so the inflated sphere is placed as the most forward surface in the velocity direction as shown in Figure 3.4.2-3. This orientation is maintained as the Orbiter moves through a variety of plasma regimes, including middle and high latitudes, the South Atlantic anomaly region, and for day/night variations. The potential of the sphere with respect to the Orbiter is maintained by a 1-kilovolt, 1-kilowatt power supply included in the instrument. During deployment, the wake characteristics are monitored by a package of diagnostic instruments situated on the second RMS so that the wake region can be spatially scanned. The instruments included in this second RMS-mounted diagnostic package are a fluxgate vector magnetometer (same unit used for the wave/particle interaction transmitter antenna), an ion mass and distribution analyzer, a planar retarding potential analyzer, a Langmuir probe, and a neutral mass spectrometer.

<u>Atmospheric Minor Constituents Profiles/Solar Flux Calibration</u> - The conduct of this experiment is essentially identical to that described for Flight 1. An ultraviolet/visible spectrometer is added to the remote sensing array of optical instruments for the purpose of measuring atmospheric limb profiles of OH and NO. This data will fill

in altitude gaps not fully covered by the other instruments. The data acquired, based on the same instrument timeline as shown for Flight 1, may be folded with the Flight 1 data to obtain more significant samples; improved signal-to-noise ratios; estimates of standard deviations of the observed constituents; and to derive correlations of constituent abundances with latitude and time, and other significant dynamical events which impact the minor constituents.

The solar flux calibration package continues to collect and build the data base for correlation with instruments on the Solar Monitor Satellite. Estimates of possible variations, due either to instrument degradation, or to real solar variations can begin to be assessed over the longer time base provided between Flights 1 and 2.

3.4.3 Flights 3, 4, and 5

For these subsequent flights the scientific objectives for each flight, as set forth by the AMPS Science Working Group, are listed in Figure 3.4.3-1. Experiments in support of each flight have been considered on a group basis for these flights rather than individual ones. A summary of the instruments and flight support equipment that is required for the experiment implementation is presented in Figure 3.4.3-2. These flights extend and continue the lines of investigations indicated for the earlier Flights 1 and 2.

The study of atmospheric minor constituents has evolved to the direct, doppler determination of atmospheric motions and dynamics, emphasizing the mesosphere/lower thermosphere. These measurements employ the Fabry-Perot interferometer and a more sophisticated, doppler-sensitive version of the laser sounder instrument. Additional minor constituents, and gaps in spatial and temporal coverage are filled by the inclusion of a solar/stellar occultation spectrometer. Finally, a gas release system is employed on the 5th flight to study basic atomic/molecular rate constants important to the chemistry and structure of the mesosphere/lower thermosphere.

These flights also continue beam-plasma studies with second and third generation experiments, using an updated electron accelerator, and an ion accelerator. These studies emphasize wave generation by particle beam injection and the mapping of electric and magnetic fields using injected beams as diagnostic probes.

Chemical release experiments employing reactive gases for conductivity modifications in the ionosphere, shaped-charge releases for magnetic and electric field diagnostics, and neutral gas releases to study anomalous ionization reactions important to solar system plasma flow processes, are all employed on these flights. They require more sophisticated versions of the gas release module employed on earlier flights, the use of shaped-charge pyrotechnic devices, and the use of a maneuvering satellite for appropriate and timely deployment.

Flight No. 3

1. Study chemical and plasma transport processes under controlled conditions by releasing chemically reactive species in the ionosphere.
2. Investigate the fundamental plasma interactions of electron and ion beams with the ambient plasma and fields, and the neutral atmosphere.
3. Investigate the coupling of VLF energy with the ambient plasma and study the associated wave-particle interactions.
4. Study the role of trace constituents in the chemistry and dynamics of the atmosphere by remote excitation and sensing, and provide diagnostic measurements for plasma/atmosphere interaction experiments.

Flight No. 4

1. Perform magnetic field line tracing experiments and measurements of parallel electric fields by means of barium releases and energetic particle beams.
2. Perform wave-wave and wave-particle interaction studies using an HF transmitter.
3. Perform fundamental studies of supersonic plasma flow around test bodies.
4. Study the role of trace constituents in the chemistry and dynamics of the atmosphere by remote excitation and sensing, and provide diagnostic measurements of plasma/atmosphere interaction experiments. (Use maneuverable sub-satellite containing diagnostic instruments in support of above objectives).

Flight No. 5

1. Investigate Alfven's critical velocity problem at low densities by releasing a suitable neutral gas (Cs?).
2. Study airglow/aurora excitation processes by injection of high density plasma streams.
3. Investigate VLF emission and propagation characteristics and study wave-particle interaction mechanisms.
4. Study the energy transfer and coupling between the ambient magnetized plasma and a large conductor at orbital speeds.
5. Study the role of trace constituents in the chemistry and dynamics of the atmosphere by remote excitation and sensing, and provide diagnostic measurements for plasma/atmosphere interaction experiments. (Use maneuverable sub-satellite containing diagnostic instruments in support of above objectives).

Figure 3.4.3-1 Mission Level Objectives

EXPERIMENTS	INSTRUMENTS AND FLIGHT SUPPORT EQUIPMENT
Active Experiments - Chemical Releases - Electron Injection - VLF Studies	New Release Mechanisms Modified or New Pressure Vessels Controllable and Uncontrollable Solid Rocket Motors Second- and Third-Generation Electron Accelerators Diagnostic Sensors Maneuverable Subsatellite 300-m Dipole VLF Transmitters Power Supply Ion Accelerator
Atmospheric Remote Sensing Experiments	Fabry-Perot Interferometer Ebert-Fastie Spectrometer Pointing Platforms Occultation Spectrometer Laser Sounder/Retroreflector Gas Physics Analysis 10-km Tether

Figure 3.4.3-2 Experiments/Instruments Summary

Additional wave experiments continue to employ the RF plasma wave sounder, and introduces a 300 meter tip-to-tip very low frequency (VLF) dipole antenna for VLF wave/particle interaction studies.

Finally a large, long-tethered, conducting balloon is introduced to investigate a number of basic plasma flow, and wave/field-aligned current generation experiments.

In all of these experiments on Flights 3, 4, and 5, the data and knowledge acquired from Flights 1 and 2 are used to enhance the definition and experimental results on these subsequent flights. Recommendations will be developed to refurbish, update and revise instrumentation and flight support equipment.

Detailed implementation plans have not been developed for these three flights. Rather, the emphasis has been placed on defining and developing the basic design drivers and operational support requirements, on the basis of the integrated experiment needs. For this purpose, a summary of the fundamental scientific capabilities of AMPS is presented in Figure 3.4.3-3. The impact of these capabilities for Orbiter/Spacelab provided support is indicated in the right column. These needs may be regarded as derived on an experiment group basis. The accommodation of the scientific instruments, and the provision of the basic support required for their efficient operation, forms the basis for payload definition and design.

Capabilities	Impact
Multiple Experiment Modes	Accommodation of Wide Variety of Instruments/ Experiments Establishment of Wide Range of Test Requirements
Remote Sensing of Multiple Parameters	Operation of Complex Battery of Instruments Accommodation of Multiple Pointing Platforms Handling of Cryogenic Instruments Contamination Control
Active Particle, Wave, Plasma Injection	EMI Control, Safety Hazard Control Boom/Antenna Accommodations
In Situ Multiple-Point Diagnostics of Particles, Fields, Waves	Definition of Low-Cost Maneuverable Satellite Deployment/Tracking/Communications of Multiple Bodies (ESPs)
Global Coverage	Communications/Data Links for Real-Time Data Interface and Control
Multiple Reuse of Instruments and FSE	Development of Rapid Refurbishment Cycle; Versatile Labcraft Accommodations Facility
Evolutionary Growth	Greater Operational Complexity; Modularized FSE
Quick-Reaction Response	Standard Interface Design, Interchangeability
Scientist/User Direct Involvement/Interaction in Experiment Operations	Interface Definition for Payload Specialist/Ground-Based Scientist in Experiment Interaction/Control

Figure 3.4.3-3 Science/Payload/Operations Requirements Summary

3.5 AMPS System Configuration Definition

Development of laboratory configurations for the AMPS flights was accomplished by combining the objectives (science and program), individual experiment requirements, instruments defined for each experiment, mission performance parameters and mission operations requirements. The prime design goal was to satisfy all experiment needs, as defined in the strawman payload definitions generated by the AMPS Science Working Group, at minimum cost. The approach to this definition emphasized establishing a performance scenario for each scientific investigation, packaging the required instruments in the payload bay, defining the support equipment for each instrument and optimizing the overall design to best fit the total job. The following paragraphs outline the Phase B study results and point out some of the features of the preliminary design.

3.5.1 Flight 1 Configuration Description

The Flight 1 configuration, as shown in Figure 3.5.1-1, consists of the interconnecting tunnel and airlock, the short Spacelab pressurized module, three Spacelab pallets, the Orbiter Remote Manipulator System (RMS) and the instrument complement as highlighted in the figure. Standard support equipment within the pressurized module, on the pallets and provided by the Orbiter are also considered as part of the configuration.

The process of establishing a design, which will accommodate each individual experiment, was initiated by defining the type of support required (i.e., pointing, electrical power, deployment of specific instruments, data types and rates, etc.), determining which requirements could be satisfied by the Orbiter/Spacelab, and defining AMPS unique support equipment. Table 3.5.1-1 summarizes the results of this analysis. The table lists, by experiment, the required instruments and the unique instrument support equipment required in addition to the significant support supplied by the Orbiter/Spacelab. It is envisioned that much of this support equipment (i.e. pointing platforms, power supplies, transmitters, receivers, etc.) will be a common requirement for many missions and therefore can be developed as Labcraft type of equipment, usable across several Spacelab payloads. A detailed list of equipment required for AMPS Flight 1 is located in Appendix A. This listing covers the basic Spacelab provisions, mission dependent (weight chargeable) equipment selected from available Spacelab and Orbiter inventory, instruments and Labcraft (flight support equipment). Total payload analysis has shown that the launch and landing weights and center-of-gravity are well within the Shuttle limitations as discussed in Section 4.1

3.5.1.1 Experiment Implementation

The significant feature of this configuration is its capability to accommodate each of the experiments as defined by the AMPS Science Working Group. The packaging of the complete payload incorporates

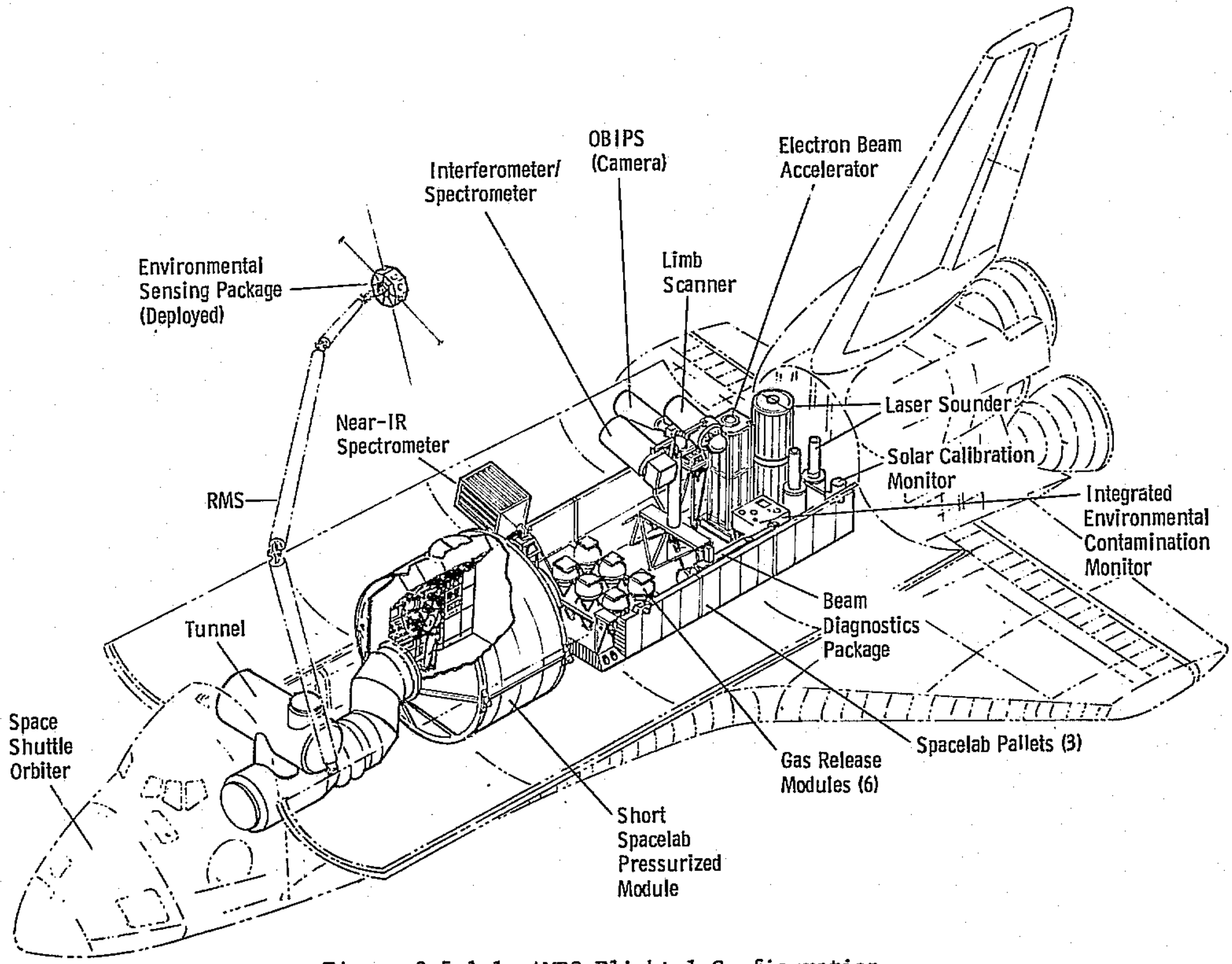


Figure 3.5.1-1 AMPS Flight 1 Configuration

Table 3.5.1-1 Flight 1 Experiment/Instrument/Support Equipment Summary

Experiment	IFRD No.	Short Title	Flight Support Equipment/Labcraft
Acoustic Gravity Wave	I-21 II-3	Gas Release Module OBIPS (L^3 TV)	Ejection Mechanism Pointing Platform Launch Locks (mpm) Installation Support Control & Display Communications & Data Handling Integrated Equipment Module Gas Release Ordnance
Electron Beam Studies	I-9 III-3 II-3 III-2 III-4	Electron Accelerator Gas Plume Release System OBIPS (L^3 TV on RMS) Vector Flux Gate Magnetometer Faraday Cup, Electrostatic Analyzer, Plasma Potential Probe	Pulse Power Supply Integrated Equipment Module Battery Power Supply Communications & Data Handling Control & Display Sensor Deployment
Atmospheric Minor Constituents Profiles	I-1 II-7 II-9 II-10	Laser Sounder Cryo-Cooled Far-IR Radiometer Near-IR Spectrometer Cryo-Cooled Interferometer/ Spectrometer	Pulse Power Supply Pointing Platforms (2) Installation Support Thermal Canister Launch Locks (mpm)
EMI Field Mapping and Space Shuttle Wake Measurements	III-25 III-18 III-23	E-Field Power Spectral Density Analyzer B-Field Antenna and Receiver Langmuir Probe Ion Mass Spectrometer Vector Magnetometer Planar RPA Neutral Mass Spectrometer	Integrated Equipment Module Communications & Data Handling Battery Power Supply Sensor Deployment Spin Table Installation Support Ejection Mechanism
Particulate and Gas Effluence	NA	Integrated Environmental Contamination Monitor	Installation Support
Solar Flux Calibration	IV-1	Array of Spectrophotometers (300 to 3500 A)	Installation Support

ample field-of-view for each instrument, modular pallet buildup to enhance ground checkout, use of the RMS to reduce design complexity and cost for required instrument deployment, equipment mounting to enhance payload center-of-gravity location, and grouping of deployed measurements to promote common usage of power supplies and data handling equipment. Safety of the vehicle and the flight crew was a prime consideration in the definition of the payload. Section 3.4.1 describes how each of the experiments are accommodated using the defined configuration.

3.5.1.2 Subsystem Features

The definition of each of the subsystems that make up the laboratory configuration was based on the guidelines as listed in Section 3.1; and the preliminary design integrated specific instrument support requirements into an optimized payload configuration. Section 4 addresses each of the subsystems in detail and the following paragraphs summarize the significant features in order to provide a capsule view of the overall laboratory configuration.

Structures and Mechanisms Subsystem - Installation of equipment, instruments and Labcraft is accommodated by the Spacelab provided pressurized module equipment racks and three meter pallets. Control and Display equipment chassis attach to provided rack slides. All pallet mounted equipment interfaces at the provided hard points. Each three meter pallet segment has been designed as an individual unit to support independent integration and transport. Equipment arrangement was based on field-of-view and payload center-of-gravity requirements. For example, the heavier instruments, permanently attached to the pallet, were located on the aft pallet. The portions of the payload which were ejected in orbit were located in the forward pallet. Equipment mounting hardware has been designed to interface with pallet hardpoints and, when required, position the instrument for optimum operation. Weight estimates for installation support hardware (trusses, brackets, platforms) were based on high design margins to reduce structural testing and related costs.

Several types of mechanical devices are provided to support instrument and mission requirements. For example, a capture/release mechanism is used for mounting the Beam Diagnostics Package. This device secures the assembly during launch, releases for deployment via the RMS and is locked after restowage of the assembly for return with the payload. A pneumatic (ordnance released) device is used to eject the gas release modules with the proper delta velocity to position the gas cloud over a selected ground station. Other types of mechanisms envisioned include: springloaded (ordnance released) ejection for the ESP, antenna and sensor deployment for the ESP and Beam Diagnostics Package, emergency ejection for equipment which is deployed beyond the Orbiter payload envelope, and a spin system to rotate the ESP.

Thermal Control Subsystem - Two primary approaches were used for definition of the thermal control subsystem: maximum use of the Spacelab provided capability, and low α/ε external coatings to cold bias equipment using heaters to control low end temperatures. Cooling for equipment located in the pressurized module is provided by the Spacelab Avionics air loop. Pallet fixed mounted equipment, such as the Laser Sounder, Electron Accelerator, IECM and the AMPS RF terminal, are coupled to Spacelab fluid loop cold plates for temperature control. Multilayer insulation thermal blankets are provided for each pallet to

simplify thermal control for components that do not require visible access to space and to minimize temperature variations within the pallets. Three methods of thermal control are provided for components extending beyond the thermal curtain: 1) cold biasing equipment using multilayer insulation and heaters, 2) thermal canister for instruments which require fine temperature control, and 3) open cycle cryogenic cooling for instruments requiring extremely low temperatures (assumed as an integral part of instrument design).

Electrical Power and Distribution Subsystem (EPDS) - Prime power is provided from the Orbiter fuel cells through Spacelab distribution. The 28 Vdc instrument and Labcraft requirements are met by this supplied capability. Conditioning of prime power is accomplished by: 1) Spacelab provided 400 Hz, three phase, 115/200 VAC inverters to furnish pointing platform power, 2) AMPS supplied peaking battery to support high power usage periods, and 3) high voltage pulse power/storage device to support the Laser Sounder and Electron Accelerator. Spacelab provided distribution to equipment racks and pallets is used to supply 28 Vdc and 400 Hz power to convenient interface boxes. AMPS power distributor assemblies provide individual instrument and Labcraft remote power control and protection on an individual pallet basis.

Electrical harness definition for power and signal distribution is based on integrated design for individual pallets. The major considerations for this design were: separate power, signal and ordnance harnesses with adequate separation to minimize EMI, primary harness installation on major payload segments to support preintegration prior to mounting on flight pallets, connectors at all interfaces to be made during the integration cycle and simplification of Spacelab interfaces.

Attitude and Pointing Control Subsystem - Basic pointing of instruments is supplied by the Orbiter. Instruments whose pointing accuracy and stabilization requirements fall within this capability are hard mounted to the pallet (Laser Sounder and Electron Accelerator). Multiple simultaneous orientation and fine pointing control is provided by three platforms. Attitude and rate sensing devices (fixed head star trackers and rate gyro packages) are provided for target acquisition, fine pointing control and platform stabilization. Use of NASA provided multi-usage Labcraft platforms, such as the Small Instrument Pointing System (SIPS) and Miniaturized Pointing Mount (MPM), is envisioned as well as NASA standard low cost attitude and rate sensors.

Data Management Subsystem - Pallet mounted instrument data handling capability is grouped into three categories: low and medium rate digital housekeeping and science data, high rate digital science data, and analog/video data. The low rate digital data interfaces with the Spacelab experiment computer through Spacelab provided Remote Acquisition Units (RAU). This computer processes the data for onboard display at the Control and Display station and routes the data for recording or ground transmission. All instrument commands are routed through the Spacelab experiment computer and RAUs. Commands can be implemented

via the keyboard at the control and display station or by ground control via the Orbiter communication system. High rate digital data is routed to the Spacelab provided high rate multiplexer which combines the incoming data from different sources and routes it for recording on the Spacelab digital recorder or for ground transmission. Analog data handling consists of AMPS unique processing equipment. Pulse signals are routed, via a switching panel, to a transient recorder to condition waveform for display, to an oscilloscope for inflight data evaluation to an analog recorder, or to the Orbiter for ground transmission. FM demultiplexing equipment is provided to recover analog data from deployed packages for onboard display. Video data can be displayed on a monitor in the Spacelab module, recorded for playback or tape return, or routed for ground transmission.

Data handling for deployed modules consists of PCM programmers, and subcarrier oscillators necessary to condition the raw instrument data for efficient transmission. Command decoders are provided for remote instrument control from the Spacelab. Each module is envisioned as independent and interfacing with the Spacelab data management subsystem through an AMPS provided RF link.

Control and Display Subsystem - The Spacelab provided cathode ray tube display, alphanumeric keyboard and function control keyboard have been baselined for the majority of control and display tasks. Interfacing with the instruments and Labcraft equipment is accomplished through the data bus as discussed in the previous section. Dedicated control and display panels are provided for safety critical functions, such as ordnance arm and safe, for functions which are inherently manual in nature, such as TV camera focus/zoom control, pointing platform tracking control for viewing chemical or gas clouds and for time displays such as GMT, orbit phases and event timing. An Orbiter provided control station is used to support experiments which require the RMS to deploy instruments. The capability for using ground control of instrument/experiment operation is also included in the configuration. The subsystem provides the flexibility to alter the mix of ground and onboard control to meet mission requirements.

Communication Subsystem - All payload data and command transmission to and from the ground is supplied via the Orbiter K_u Band transmitter and the TDRSS. Digital data up to 50 Mbps is routed to the Orbiter via the Spacelab high rate multiplexer for real time transmission, or can be transmitted in delayed time from the Spacelab digital recorder. Analog and video data can be routed directly for transmission or recorded on AMPS provided recorders for delayed time transmission. A limited backup communication capability is supplied via S-band links with the STDN. No AMPS unique hardware is required to support the provided capability for air-to-ground communication.

Communication with RMS deployed or free flying modules is provided by an AMPS unique RF link. The pallet mounted RF terminal, which is common for all deployed packages, consists of an S-band command

transmitter, receivers, RF multiplexer and antenna capable of receiving digital/video/analog data and transmitting commands. The RF terminal interface with the payload data acquisition is similar to the interface with onboard instruments as described for the data management subsystem. Each deployed package is configured with an RF system selected to accommodate the instrument requirements. Presently defined requirements allow for low cost design by selection of off-the-shelf subassemblies qualified to near expected environments. Commonality of components between deployed modules and the pallet terminal was also a major consideration.

Deployed Instrument Support Subsystem - Evaluation of the many requirements for remote measurements in support of AMPS investigations led to some grouping of closely related instruments into a single package. Grouping criteria included: type and location of measurement, sequence of measurements, free flying vs RMS deployed positioning, minimization of support equipment (communication, power supply, support structure, etc.) and overall subsystem cost. The recommended configuration for Flight 1 consists of:

(1) Gas release module (6 required) - High pressure gas bottle (assumed instrument provided) and necessary support equipment to achieve gas release at the proper orbital position.

(2) Beam Diagnostics Package - A series of instruments capable of measuring, while exposed to the electron beam, characteristics such as current density, particle acceleration energies and beam plasma potential. The structure acts as an extension to the RMS in order to position the instruments over the electron accelerator. A low light level TV camera visually monitors the electron beam structure at various distances from the accelerator. A vector magnetometer is provided to sense earth's magnetic field direction.

(3) Environmental Sensing Package - A combination of instruments capable of measuring electromagnetic interference levels and Orbiter wake characteristics at various positions with respect to the Orbiter payload bay when positioned by the RMS. Measurements at greater distances from the Orbiter are accommodated during a free-flying mode after ejection from the RMS. A vector magnetometer is provided to sense earth's magnetic field direction.

3.5.2 Flight 2 Configuration Description

The Flight 2 configuration, as shown in Figure 3.5.2-1, is similar to that for Flight 1. The significant differences are: the addition of a second RMS to support plasma wake studies, a modified instrument complement as shown in the figure and four new integrated

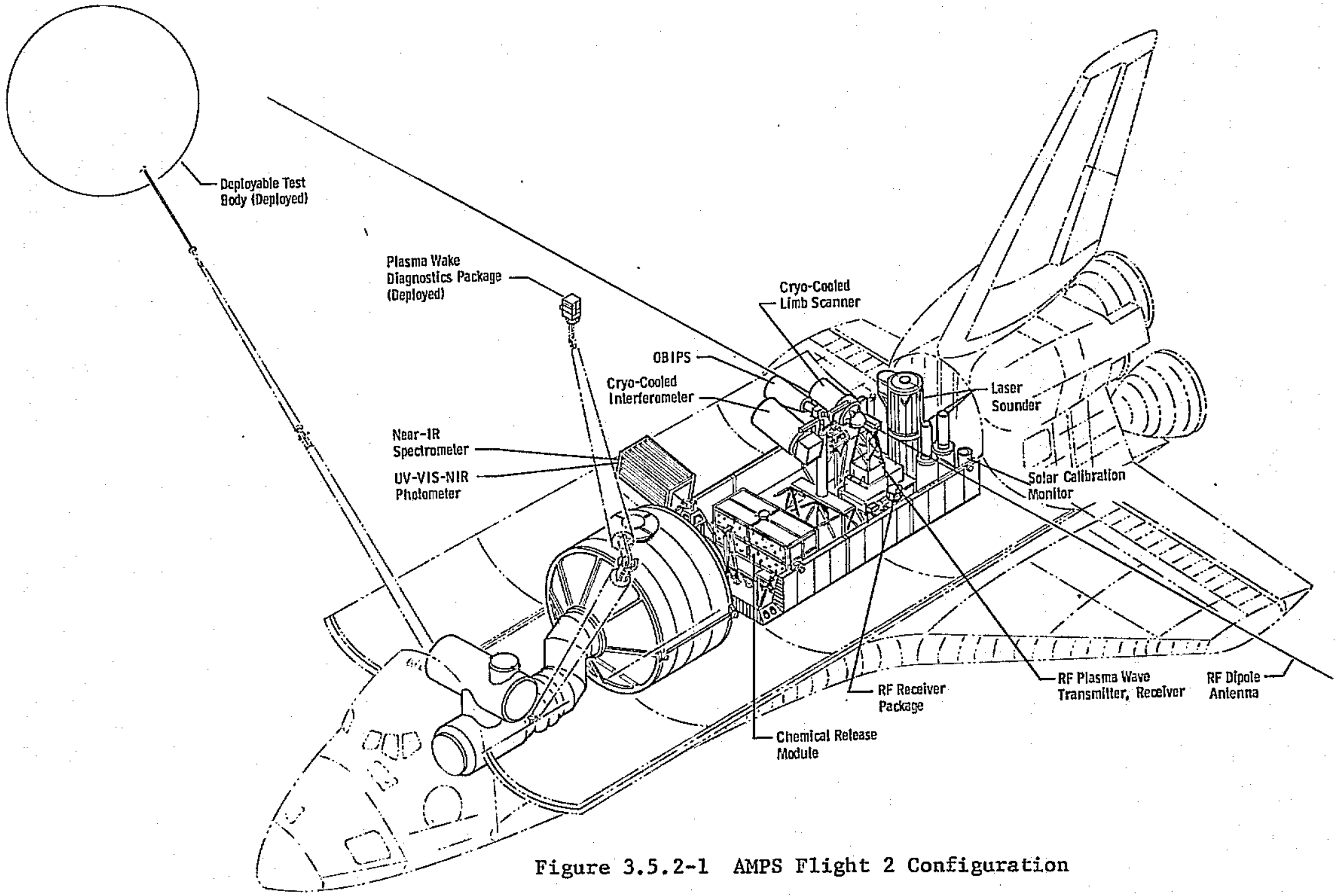


Figure 3.5.2-1 AMPS Flight 2 Configuration

equipment modules for deployed measurement support. Maximum use of Flight 1 hardware was emphasized to promote the concept of a flexible laboratory capable of sequential evolution over a long term experimental program.

The experiment/instrument/flight support equipment relationship is portrayed in Table 3.5.2-1. Of particular interest, is the percentage of the support equipment which is carried over from Flight 1 directly or with some modification. The high percentage of required support provided by the Spacelab/Orbiter remains essentially the same as for Flight 1. Section 3.4.2 describes the use of this defined configuration in the performance of each of the experiments.

3.5.2.1 Subsystem Features

Subsystem configurations for Flight 2 were developed to promote maximum usage of Flight 1 hardware. Therefore, a significant percentage of the design approaches and features are identical for this configuration. The changes are defined in detail in Section 4 and the significant features outlined below.

Structures and Mechanisms - The installation concept remains the same with only minor redesign to replace the electron accelerator by the RF plasma wave package. New designs for four new integrated modules are included. Reusable installation support trusses, in adduition to those used to mount instruments reflown for Flight 2, are: platform on which the chemical release module is mounted and the truss/platforms for the Plasma Generator and Plasma Wake diagnostic package. Capture release mechanisms are also reusable for these two modules.

Data Management and Communication Subsystems - Both subsystems remain essentially the same except for communications with the deployed packages. One of the wideband receivers in the pallet RF terminal must be replaced with a narrowband receiver. Each of the four deployed modules is outfitted with an RF system and data acquisition system to support its operation. Commonality of design between the modules of both flights was emphasized in order to reduce costs.

Deployed Instrument Support Subsystem - Instruments were again grouped by type and location of measurements as discussed for Flight 1 in Section 3.5.1. The recommended configuration for Flight 2 consists of:

(1) Chemical Release Module - 64 barium thermite canisters packaged to exhaust outboard during the burn cycle. Equipment is provided to time-sequence the ignition of canisters over a period which will produce a 100 Km cloud. An RMS and effector is also provided to support removal from the payload bay.

(2) Plasma Wake Generator - Package containing deployable test

Table 3.5.2-1 Flight 2 Experiment/Instrument/Support Equipment Summary

Experiment	Instruments		Flight Support Equipment/Labcraft
	IFRD No.	Short Title	
Conductivity Modification	I-21 II-3 II-4	Checmical Release Module OBIPS (L^3 TV) UV Spectrometers	Integrated Equipment Module Communications & Data Handling • Display & Control • Installation Support • Pointing Platforms (2) • Launch Locks (mpm) •
Wave Particle Interactions	I-12 III-2	RF Plasma Wave Package Flux Gate Vector Magnetometer	Installation Support •
Long-Delay Echo	I-12 III-2 I-12 III-2	RF Plasma Wave Package Flux Gate Vector Magnetometer Subsatellite RF Plasma Wave Receiver Subsatellite Flux Gate Vector Magnetometer	Integrated Equipment Module Battery Power Supply Communications & Data Handling • Sensor Deployment • Installation Support • Ejection Mechanism • Spin Table
Plasma Flow	III-17 III-2 III-10 III-18 III-22 III-23	Deployable Test Body (Spere) Flux Gate Vector Magnetometer Ion Mass Distribution Analyzer Planar RPA Langmuir-Type Current Collector Neutral Mass Spectrometer	Integrated Equipment Modules (2) Communications & Data Handling • Battery Power Supplies Installation Support • Extension Boom
Atmospheric Minor Constituents Profiles	I-1 II-7 II-9 II-10 II-4	Laser Sounder Cryo-Cooled Far-IR Radiometer Near-IR Spectrometer Cryo-Cooled Interferometer/ Spectrometer UV-Visible-Near-IR Spectrometer/ Photometer	Pulse Power Supply • Pointing Platforms (2) • Installation Support • Thermal Canister • Launch Locks (mpm) •
Solar Flux Calibration	IV-1	Array of Spectrophotometers (300 to 3500 Å)	Installation Support •

body, power supply to vary potential with respect to the pallet, and data handling equipment. An extendable boom is also supplied to position the sphere an additional 15 ft. from the payload bay. Capability is provided to inflate the test body after deployment by the RMS and to eject it prior to restowage of the module on the pallet.

(3) Plasma Wake Diagnostics Package - A series of instruments capable of measuring the characteristics of the wake generated by the deployable test body. The RMS will position the module both in the wake and the ambient plasma for comparison

data. A vector magnetometer is supplied to sense earth's magnetic field direction.

(4) RF Receiver Package - The RF receiver (instrument provided and matched to the pallet mounted transmitter) is packaged along with the necessary support equipment (data handling, power, command). Capability is provided to eject the package at approximately 5 m/sec delta velocity, spin the package for stability control, and deploy instrument antennas after ejection. A vector magnetometer is provided to sense earth's magnetic field direction.

3.5.3 Follow-On Missions

The objectives of evaluating the follow-on AMPS missions, Flights 3, 4, and 5, were twofold: (a) to determine what the support subsystem design drivers are and (b) to incorporate any design changes in the initial Flights 1 and 2 configurations that would enhance downstream evolution. Section 3.4.3 describes the science objectives defined by the AMPS science working group and outlines the types of experiments which be performed during these flights. The first step in the evaluation process was to develop top level experiment operation descriptions for the major new or changed investigations. From these data, specific experiment requirements were defined, such as: type of instruments, significant changes in pointing and stabilization needs, sensor location (remote vs fixed pallet) and chemical/gas release configurations.

The next step was to define the subsystem support design drivers by determining the type of subsystem support required to satisfy these experiment requirements and promote full accomplishment of the investigations. The configurations developed for Flights 1 and 2 were used as baseline for subsystem support capability. Table 3.5.3-1 presents a summary of the evaluation results. The first column lists the significant new or changed experiments envisioned. The second and third columns list the design drivers and potential configuration changes respectively. In addition to these specific drivers, the analysis indicated that overall space for mounting instruments (with proper field-of-view and gimbal clearance) and payload weight limitations would necessitate prioritizing experiments and instruments as the laboratory is developed. Some examples of the impacts are discussed below.

Chemical Releases - Chemical release requirements for follow-on missions vary from significant altitude and distance changes with respect to the Orbiter to specific orientation requirements at the point of release. The distance/position requirements indicate a need for a special thrust motor and an inherent attitude control capability in order to transport the release. Shaped charges require that each release be oriented with respect to the magnetic field lines prior to release which indicates the need for a maneuverable subsatellite. Combination of the two requirements could result in providing a maneuverable subsatellite with variable thrust capability to both transport

Table 3.5.3-1 Follow-on Flight Configuration Impacts

Experiment	Design Driver	Potential Configuration Change
Flight 3		
Chemical Release	Release Altitude	Move Orbiter to Release Position or Thrust Motor & Attitude Control for Release Module
VLF Antenna Studies	Long Antenna Dynamics	Dipole and Loop Antennas
	Radiation & Propagation Measurements	May Require Maneuverable Subsatellite
Atmospheric Remote Sensing	Add Instruments to be Pointed	Reconfigure Pointing Platforms Reduce Instruments
Flight 4		
Chemical Release	Shaped-Charge Requirements	Thrust Motor & Attitude Control for Module
Plasma Flow	Position & Shape of Deployed Body	Free Flying Test Body Maneuverable Subsatellite & Tracking/Communication Antenna
Atmospheric Remote Sensing	Add Instruments to be Pointed LIDAR Reflective Subsatellite	Reconfigure Pointing Platforms Gimbal Laser Sounder
Flight 5		
Chemical Release	Shaped-Charge Requirements	Thrust Motor & Attitude Control for Module
Plasma & Ion Accelerators	New Type Hardware	Reconfigure Pallet Modify HV Power Supply
Field Line Generator	Long-Tethered Balloon System	10K Tether/Winch and Diagnostic Subsatellite
Atmospheric Remote Sensing	Same as Flights 3 & 4	Same as Flights 3 & 4

and orient the releases. Integration of such a subsatellite presents some problem because of size and weight. Mounting over the Spacelab transfer tunnel is a viable approach within center-of-gravity and payload weight limitations

<u>Pointing Platform Configuration</u> - The bulk of changed pointing requirements indicate a need to add instruments to the platforms provided for the initial flights with some repackaging required. Development of standardized mounting bracketry and thermal control canisters is indicated. A major impact is foreseen in order to accommodate LIDAR reflective experiments because of the need to gimbal the instrument rather than fixed mounting to the pallet. Space to mount the combined instrument/gimbal, including sufficient gimbal freedom, as well as the added gimbal weight will require significant reconfiguration.

<u>Very Low Frequency Antenna Studies</u> - Very long antennas, required to provide efficient low-frequency transmission, are foreseen as design/development drivers. The inherent low stiffness characteristics of this type of antenna impacts the requirement to orient the antenna while extended. Alternative approaches to experimental performance require investigation; such as, using a maneuverable satellite to deploy a wire antenna or specific limitations to Orbiter reorientation while the antenna is deployed.

<u>Impact of Follow-On Missions on Initial Configurations</u> - Although the major impact of follow-on missions will be the need to select a portion of the instruments because of weight and space limitations, this analysis did influence initial configuration definition. Examples of this influence are outlined in the following paragraphs.

(1) Use of RF link for deployed instrument data recovery and command provides a capability required for the maneuverable subsatellites foreseen for downstream missions. Design of the pallet located RF terminal and its interface with the onboard data management subsystem was developed to support this future requirement.

(2) Selection of pointing platforms was based on a wide variety of instrument weights, envelopes, pointing direction and thermal conditioning requirements. Simultaneous pointing and the need to incorporate small instrument/platform combinations with fixed mounted payloads also influenced the configuration. Platforms capable of handling all ranges of instruments were considered necessary to satisfy all these requirements and provide necessary flexibility for the AMPS laboratory.

(3) The investigations defined for the follow-on flights require the addition of a maneuverable vehicle to position instruments and chemical releases. Since this capability must be supplied eventually, rescheduling the development of a maneuverable subsatellite so as to support the initial flights is attractive. Section 5.9 presents an evaluation of the impact of supplying this capability early in the AMPS program.

4. SUBSYSTEM DEFINITION

4.1 Structures and Mechanisms Subsystems

The structures and mechanisms subsystem study effort encompasses the packaging of the Spacelab and AMPS payload components within the Orbiter payload bay, while remaining within the Orbiter and Shuttle constraints. The goal of this effort was to prove the physical feasibility of the strawman payloads. That feasibility was established while identifying the major constraints of Orbiter/pallet allowable interface locations, Orbiter allowable longitudinal C.G. location, pallet train payload capability, pallet volume available, and pallet hardpoint locations.

The Spacelab facilities provide the primary support structure for the AMPS payload in the Orbiter bay. The pressurized module provides the capability for the "man in the loop" interaction with the experiments. The Spacelab pallets play a major role in the AMPS structural system. They allow the support of large, heavy equipment by designing only intermediate or secondary structure between the equipment and the pallet. The use of multimission support equipment (MMSE) and other "off the shelf" hardware has been maximized. The miniaturized pointing mount, small instrument pointing system, environmental canister, and a capture/release mechanism are in this category.

Mechanism requirements defined include capture/release, planned ejection, launch/landing locks, and emergency jettison devices. Where needed, AMPS support structural design provides for equipment mounting, FOV, and interface with the pallets. Preliminary design was accomplished on Integrated Equipment Modules to package instruments and support equipment when necessary to meet experiment operational requirements.

4.1.1 Structures and Mechanisms Requirements

The preliminary steps in the development of the structures and mechanisms designs involved the determination of the basic science requirements from the experiment objectives and then refining these to obtain specific structural and mechanism design requirements. The more detailed design requirements evolve from the basic design concept as it is developed and integrated with the total payload requirements.

The experiments to be performed on Flight 1 include: acoustic gravity waves, electron beam studies, minor constituents, and EMI field mapping. Table 4.1-1 defines the basic science requirements needed to support the science activity. In addition, first level structural and mechanical requirements such as pointing, orientation, alignment, and operational interfaces are listed.

Table 4.1-1 Structures/Mechanisms Requirements as Derived from Science Requirements

EXPERIMENT	INSTRUMENT	SCIENCE REQUIREMENTS	STRUCTURAL REQUIREMENTS	MECHANISM REQUIREMENTS
Acoustic Gravity Waves	I-21	Release gas cloud away from orbiter.	o Package gas & subsystem for ejection.	o Eject from orbiter, remote nonpropulsive gas release.
	II-3	Tracking & viewing of gas cloud.	o Mount on pointing platform	
Electron Beam Studies	I-9	Generate electron beam	o Mount/orient accelerator parallel to Z axis.	
	III-3	Gas plume release into beam	o Locate on accelerator	
	II 3	Observe beam/field interactions.		o Deploy away from orbiter for maximum view of beam.
	III-2	Measure beam/magnetic field.		o Deploy/position over beam.
	III-4	Map beam cross section		o Deploy/position over beam.
Minor Constituents	I 1	Transmit/receive backscatter laser beam.	o Mount/orient parallel to Z axis.	o Coalign transmitter & receiver.
	II-7	Measure spatial distribution of infrared radiation of earth's limb (cryo cooled, works with II-10).	o Mount on pointing platform (provide cryo storage)	o Coalign with II-10
	II-9	Obtain spectra of atmosphere (sun & earth pointing).	o Mount on pointing platform (view sunrise/sunset & earth limb).	
	II-10	Measure atmospheric emission in infrared spectrum (cryo coded spectrometer, works with II-7).	o Mount on pointing platform (provide cryo storage).	o Coalign with II-7
EMI Field Mapping & Shuttle Wake Measurements.	III-18	Measure electron energy distribution in the unperturbed area away from orbiter.		o Deploy from orbiter (near field). Eject from orbiter (far field).
	III-23	In-situ neutral gas composition measurements in unperturbed atmosphere.		o Deploy from orbiter (near field). Eject from orbiter (far field).
	III-25	Measure EMI levels & plasma pertu[illegible]tions in the orbiter [illegible]icinity.		o Deploy from orbiter (near field). Eject from orbiter (far field).

4.1.2 Structures and Mechanisms Concepts (Flight 1)

The structural and mechanisms design approach was to develop a payload that met the science objectives, met the Orbiter and Spacelab constraints, utilized the maximum amount of MMSE and Spacelab provided equipment, and provided low cost, reusable AMPS unique support equipment. The Flight 1 preliminary design arrangement developed to meet these goals is described in Figure 4.1-1.

The AMPS payload configuration is made up of the interconnecting tunnel and airlock, the remote manipulator system, the small Spacelab module, three Spacelab pallets, the AMPS instruments, multimission support equipment, and AMPS unique support equipment. Orbiter furnished equipment in the cargo bay includes the tunnel adapter, airlock, tunnel

4. SUBSYSTEM DEFINITION

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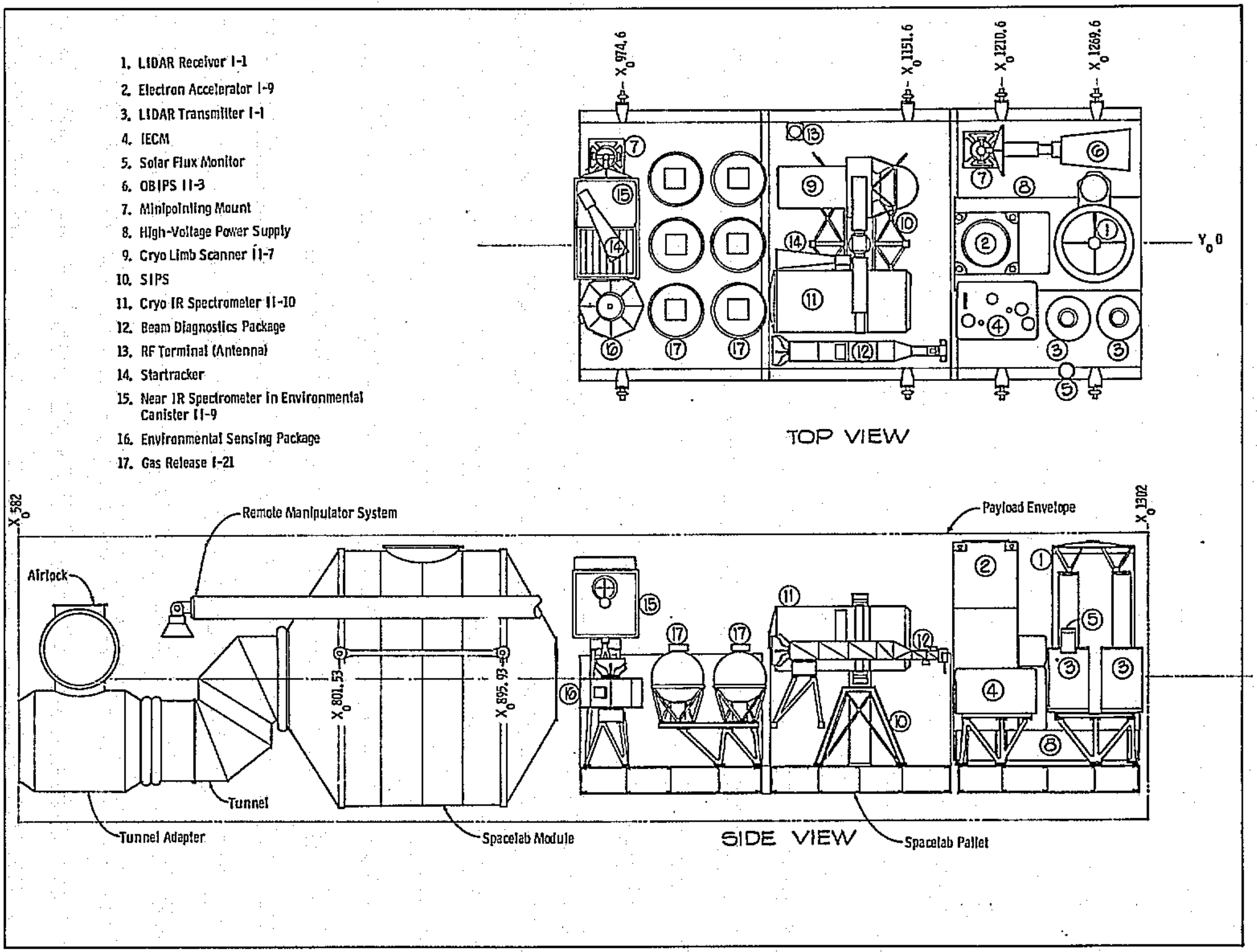


Figure 4.1-1 AMPS Flight 1 Physical Configuration

and the remote manipulator system. The tunnel system provides for transfer of crew and equipment between the Orbiter crew compartment and the pressurized module. Use of an 11.665 inch spacer segment plus a double loop flex section provides the proper tunnel length to match the AMPS module position. The RMS on Flight 1 is used to deploy the beam diagnostics package and the ESP module outside the payload bay for experiment measurements and viewing. The RMS is installed along the left sill of the Orbiter and in its stowed position extends from forward of the module to within 17.2 inches (.44 metres) of the end of the pallet train. A configuration consisting of the Spacelab small module and three Spacelab pallets was chosen to satisfy the goal of maximum usage of existing equipment. The pressurized module provides a laboratory area for conduct of the scientific investigations. The pallets provide the primary support for the AMPS instruments and support equipment. Location of the module and pallets at the aft end of the payload bay was an attempt to alleviate the longitudinal C.G. problem. The pallet train has been positioned as far aft as possible where payload attach points exist that are compatible with the pallet fitting requirements. As located, 5.3 inches (.13 metres) clearance exists between the aft pallet and Station 1302. A pallet train configuration of two joined pallets plus a separate aft pallet was selected to achieve greater load capability. The standard three linked pallet arrangement has a total capability of 11,023 pounds (5000 kg). By using a 2P + 1P configuration, the capability is increased to 17,897 pounds (8118 kg). The weight penalty for the extra pallet keel and longeron fittings required in this scheme is 434 pounds (197 kg). The overall length of the 2P + 1P layout is 349.2 inches (8.87 metres), which is identical to the three pallet train due to the pallet splice configuration. The pressurized module is located approximately 15 inches (.38 metres) forward of the first pallet. This module and pallet layout does not require utility bridges between the module and the pallets or between pallets due to the small gaps.

There are seventeen different instruments used on the AMPS Flight 1. They range in weight from 1102 pounds (500 kg) each for the cyro cooled instruments (II-7 and II-10) to 6.6 pounds (3 kg) for the solar flux monitor (IV-1). Within the bay, instruments are mounted separately, supported on pointing platforms, and grouped together as deployed instrument modules. A total of twenty separate instrument packages or modules required mounting on the Flight 1 configuration. Physical characteristics of the individual AMPS instruments are described in the IFRDs and Appendix B. Instruments have been positioned on the pallet to satisfy the experiment/instrument constraints and fit within the Shuttle packaging constraints. Section 5.2.1 describes the study approach and rationale utilized for this positioning task. Key features used in the instrument location task included; weight, size, field of view, and operational requirements. The most significant design constraint encountered was the Orbiter longitudinal C.G. requirement. This resulted in starting the payload layout by placing the heaviest items on the aft pallet, and then working the center and forward pallets with decreasing weight items while satisfying the operational and physical

constraints. Multimission support equipment (MMSE) was integrated into the layout wherever possible to reduce design and procurement costs and provide maximum reusability. MMSE included on the Flight 1 configuration included; minaturized pointing mount (MPM), small instrument pointing system (SIPS), and an environmental canister. Section 5.2.1 discusses the selection of the pointing platforms to satisfy the pointing requirements of OBIPS, the cryo limb scanner, the cryo IR spectrometer and the near IR spectrometer. An environmental control system was needed for the near IR spectrometer to satisfy the instrument thermal requirements. The MSFC MMSE environmental canister was selected because the instrument and canister were physically compatible and the canister was designed for use with the MPM.

Instrument Support Design - Design concepts investigated for the support of instruments included the following structure types; platform, truss, and direct. The platform structural approach uses a grid like structure that spans the pallet above the floor and is supported from pallet hardpoints. Truss type support structure consists of tubular members which extend from the instrument to pallet hardpoints. Direct mounting techniques involve the use of brackets or fittings which directly mount an instrument to the pallet hardpoints. Table 4.1-2 lists the advantages and disadvantages of the three types of structure.

Table 4.1-2 Comparison of Structural Mounting Concepts

PLATFORM	o not constrained by hardpoint locations. o increased flat surface area o can vary platform elevation	o flat plate type construction would have relatively low stiffness. o loss of pallet usable volume o not compatible with equipment requiring different mounting levels.
TRUSS	o stiff, efficient load paths o multiple attach to hardpoints possible. o good strength to weight ratio	o shared hardpoints (if utilized) present integration and interface problems. o not normally adaptable for reuse.
DIRECT	o good load paths o lightweight	o limited attachments (hardpoints) o viewing requirements often mean structure extension.

Use of all three types of support structure can be found on the Flight 1 layout. However, there are more of the truss type structures utilized. The good strength to weight ratio, low cost fabrication techniques, and the need to mount equipment at varying elevations were the prime reasons for choosing the truss approach. Platform structure was used on the forward pallet to mount the gas release experiment. Here a large surface area was required for supporting the gas release modules. Direct mounting brackets were used in a number of places including the pulse power supply on the aft pallet. In general, the support structure material selection approach, was to use 6061 aluminum alloy wherever possible because of availability, ease of fabrication, and stress corrosion resistance.

Rather than attempt a description of all of the support structure on the Flight 1 layout, key features of particular support structures on the three pallets will be discussed. The aft pallet is the heaviest and most complex pallet. Instruments mounted on this pallet are the electron accelerator with pulse power supply, LIDAR receiver, two LIDAR transmitters, OBIPS on the MPM, IECM, and the solar flux monitor. In addition, a peaking battery for the pulse power supply, a dedicated heat exchanger for the LIDAR transmitters, seven cold plates, and the unique support structure occupy this pallet. Sketch SK05-011 in Appendix C shows the aft pallet structural layout. The packing density on this pallet is high as a result of attempting to get as much weight on the pallet as possible. As full as the pallet appears, only 40% of the individual pallet weight capacity has been used. Interfaces on this pallet include instrument to support structure and MMSE (MPM), instrument to instrument, and support equipment (structure and components) to pallet. There are no direct instrument to pallet interfaces (i.e., all instruments require mounting structure). Of interest on this pallet, are the mounting structures for the pulse power supply and the LIDAR receiver. These instruments are used as examples of direct mounting and truss mounting structures. The major item to be supported is the pulse power supply. This 1323 pound (600 kg) unit is supported 5 inches (.13 metres) above the pallet floor by eight machined bathtub brackets (direct mounting). The brackets attach to the pallet hardpoints located along both edges of the pallet floor. Clearance above the floor is provided for cold plate and fluid line attachments. Because of the number of items located on the pallet, sharing of hardpoints was required. To facilitate access and reduce integration complexity, the pulse power supply brackets are used for attachment points for support structures from OBIPS (MPM), IECM, and LIDAR. The LIDAR receiver uses a C.G. mounting arrangement that is supported by a truss type structure consisting of six struts. Four struts are attached to the pulse power supply brackets and two struts attach to the aft pallet to Orbiter fitting. The use of the pallet to Orbiter fitting as a structural support attach point was required to satisfy a need for a $\pm Y$ load reaction point and provide clearance with the LIDAR transmitters. The receiver is stabilized by the use of brackets between the base of the receiver and the top of the pulse power supply. Instrument support structure on the remainder of

the pallet consists of: truss mounting for OBIPS, the IECM, and the LIDAR transmitters; instrument to instrument attachment for the electron accelerator; and a sheet metal bracket support attached to the pallet panel for the solar flux monitor. The peaking battery and dedicated heat exchanger make up the Labcraft located on the aft panel. A cold plate in the standard location supports the peaking battery and the heat exchanger is mounted directly to the pallet panels. At 83 pounds (37.7 kg), the heat exchanger is possibly too heavy for panel mounting. A review of the panel insert capability, when available, will be used to determine the feasibility of the proposed mounting. As an alternate mounting scheme, the exchanger could be attached to the bottom of the LIDAR transmitter support structure.

The center pallet is dominated by the cryo limb scanner and cryo IR spectrometer mounted on the SIPS. The beam diagnostics package and the RF terminal package (support equipment) are also located on this pallet. The SIPS base structure is designed to interface directly with the pallet hardpoints. AMPS unique support structure is required to adapt the cyro instruments to the pointing system. A machined frame similar to the SIPS canister gimbal frame is proposed as the attachment to the yoke. These mounting frames require adjustment capability to allow boresighting of the cryo instruments. On orbit alignment will also be possible by using a sensor system between the two instruments. Support structure near the sill at the front and rear of the pallet is used to position the beam diagnostics package for RMS access. The truss type structure supports the beam diagnostics capture/release mechanisms and is in turn supported from the pallet hardpoints. A shared hardpoint interface is required between the beam diagnostics package support structure and the SIPS side struts. A modification or new SIPS strut is needed to accommodate this shared interface. The capture/release mechanism as discussed in Section 5.2.2 allows the beam diagnostics package to be released from its launch support position for deployment and then reattached for landing. A standard cold plate on the right side of the pallet provides the support for the RF terminal electronics package. A sheet metal bracket attached to the pallet panel near the sill mounts the RF conical antenna.

On the forward pallet are the six modules for the gas release experiment, the near IR spectrometer in the environmental canister on the MPM, and the environmental sensing package. The gas release modules are mounted on a platform that spans the pallet about 27 inches (.68 metres) above the pallet floor. This welded frame platform provides the surface area required to mount the 34 inch diameter spheres (.86 metres). The platform design is one of the more challenging design tasks due to the high weight (2840 pounds, 1289 kg) and cantilever configuration. This design again points out the restrictions on instrument location and support structure design that result from the hardpoint locations on the pallet. Alternate design approaches available to eliminate the cantilever, if required, include providing struts forward to the hardpoint locations now used by the MPM and adding hardpoints at the center of the

pallet. The ESP is attached to a capture/release mechanism that is similar to that used on the center pallet. Tubular struts from four pallet hardpoints support the capture device and the ESP. A requirement for over the sill viewing dictated the location for the MPM above the pallet floor. A truss structure provides the support for the MPM and also supports the launch lock mechanism at the forward end of the MPM environmental canister. The requirement calling for isolation of the MPM gimbals from loads during Shuttle accelerations dictated the use of a launch lock. The launch lock system proposed consists of two main locks at the MPM/canister interface and a third lock which supports the end of the canister.

Integrated Equipment Modules - There are two deployed modules on Flight 1 that were developed to house instrument groupings for specific experiments. The ESP module was designed to support the EMI field and Shuttle wake mapping experiment. A description of the ESP module involving experiment objectives, operational modes, and a description of the module subsystems can be found in Section 4.8. Seven instrument packages and eight support equipment packages are mounted in an eight sided structural module that is 41.5 inches (1.05 metres) in diameter and 36.5 inches (.93 metres) high. A tubular frame at the periphery and at a concentric inner ring provides the basic structural frame that provides for equipment mounting to the various panels. Instruments and the power supply battery are mounted in the exterior bays while most of the support equipment is located in the central core. Shear panels are removable from the exterior faces to provide for access and maintenance. Removable access panels are also provided at the top and bottom. Diagonal framing along the radial panels adds to the structural stability of the unit. Sketch SK05-004 located in Appendix C illustrates the layout arrangement and structural concept. A spin system, separation (ΔV) system, capture/release mechanism, and sensor deployment devices are included in the mechanism designs required on this module. Spin requirements dictate a spin mode while on the RMS and a spin stabilized free flight mode. A mass properties analysis was conducted to determine the spin stability. As configured with sensors and antennas deployed, the spin principal moment of inertia ratio exceeds the required 1.1 to 1 ratio (actual 1.5) required for stable spin. The spin axis is within 2.26 degrees of the rotational axis without the addition of ballast which would be added to null the system out. Section 5.2.2 discusses the mechanism selections for the ESP and SK05-004 illustrates some of the proposed details. The ESP module weight is 307 pounds (139.5 kg), of which 143 pounds (64.9 kg) is structure and mechanisms.

The other deployed module is the beam diagnostics package which is used to perform the electron beam studies experiment. Here, the instruments are deployed and oriented to view the electron accelerator generated beam to determine beam and magnetic field interactions. Another operational mode requires insertion of the instrument group into the beam. This involves measurements within a specified parallelepiped volume above

the accelerator. As the RMS reach over the aft pallet did not satisfy the experiment requirements, a module configuration in the shape of an extension boom took form. The preliminary design concept is shown on sketch SK05-005 in Appendix C. This square boom type framework is 116 inches (2.95 metres) in length and is 12 inches (.3 metres) by 13 inches (.33 metres) at the large end and steps down to 6 inches (.15 metres) square. Welded tubular members make up the structural frame that also houses the instruments and support equipment. Diagonals are used to stabilize the truss structure except in the OBIPS and magnetometer bays where a stiffened skin panel allows for installation and access. Equipment and instrument mounting is by secondary structure or brackets supported from the main frame. Mechanisms associated with the beam diagnostics package are the capture/release device, the magnetometer sensor launch and landing lock, and the sensor deployment device. The boom length required that pallet interface support structure be provided at more than one location. Capture/release devices were positioned near the OBIPS end and near the boom tip. Use of the "standard" device which reacts X, Y and Z loads at the heavy end and use of a less complex Z reaction only device at the tip has been proposed. Power connector interface capability has been included to provide heater power and commands while stowed on the pallet. Deployment of the 1.1 pound (.5 kg) magnetometer sensor during operation necessitates both an extension device and a lock system to react launch and landing loads. A storage tubular extendible member was selected (as described in Section 5.2.2) to satisfy the 39.4 inches (1.0 metres) extension requirement. A solenoid operated pin puller arrangement is the preliminary design concept selected for the sensor lock system. EMI sensitivity of the sensor will require solenoid shielding or possible redesign for remote actuation by linkage to prevent sensor contamination. Subsystem definition and experiment operation details for the beam diagnostics package are expanded upon in Section 4.8. The structure and mechanisms account for 66.6 pounds (30.2 kg) of the 315.8 pound (143.3 kg) beam diagnostics package total weight.

4.1.3 Structures and Mechanisms Concepts (Flight 2)

The Flight 2 layout is a good example of the reuse of support structure from one flight to the next. Figure 4.1-2 illustrates the Flight 2 physical configuration. Although there are five instrument deletions and additions between Flights 1 and 2, only two support structures require major modification or replacement. The greatest impact occurs on the aft pallet where the electron accelerator and IECM are replaced by the RF plasma wave package and the RF receiver package. The RF plasma wave package is located on top of the pulse power supply in the space previously occupied by the accelerator. Attachment to the pulse power supply would require new support structure as would the mounting of the dipole antenna above the sill. Removal of the IECM provides a mounting position for the RF receiver package. Here the support structure could be modified for reuse, but the weight of the

1. LIDAR Receiver I-1
2. High-Voltage Power Supply
3. RF Plasma Wave Package I-12
4. LIDAR Transmitter I-1
5. Solar Flux Monitor IV-1
6. OBIPS II-3
7. Minipointing Mount
8. RF Receiver Package
9. Plasma Wake Diagnostics Package
10. Cryo Limb Scanner II-7
11. Cryo IR Spectrometer II-10
12. SIPS
13. RF Terminal (Antenna)
14. Startracker
15. Chemical Release I-21
16. Plasma Wake Generator III-17
17. UV-VIS-NIR Spectrometer II-9 & Two Near IR Spectrometers II-4 in Environmental Canister

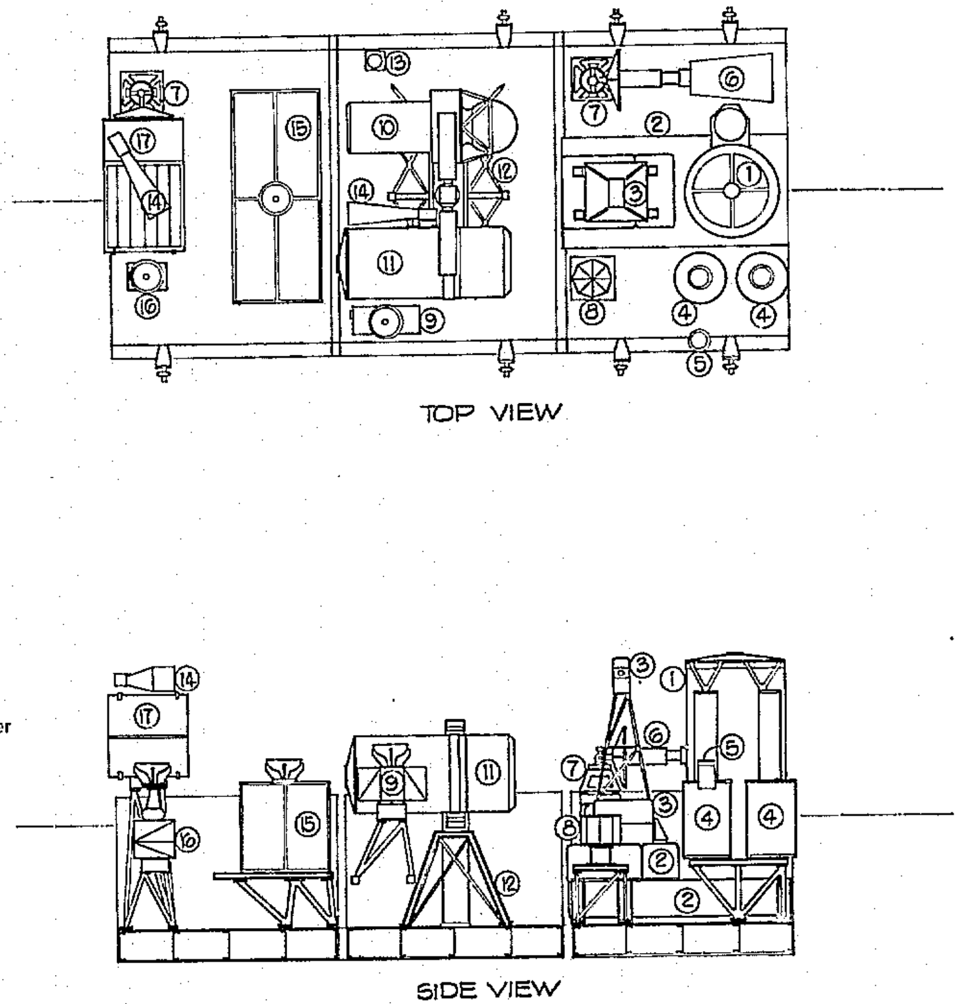


Figure 4.1-2 AMPS Flight 2 Physical Configuration

RF receiver package being 619 pounds (281 kg) lighter than the IECM did not justify this large a support structure. Mounting of the RF receiver package would use a similar truss type structure as on Flight 1. On the center pallet an exchange of deployed modules makes up the Flight 2 transition. The beam diagnostics package is removed and the plasma wake diagnostics package is supported in the same position using the main capture/release device from Flight 1. The secondary capture/release device located at the aft end of the pallet is not required for Flight 2. Deletions on the forward pallet include the ESP module and the six gas release modules. Reuse of the ESP module capture/release device and support structure for the added plasma wake generator is a direct interchange with no support equipment modifications. The chemical release will be planned to use the Flight 1 gas release support structure with modifications on top of the platform to provide a release device for deployment. As the chemical release module is 976 pounds (443 kg) heavier than the gas release modules, the support structure design for Flight 1 would have to be overdesigned accordingly. Another change on the forward pallet involves the MPM canister. The addition on the MPM of two UV-Visible-near IR spectrometers to the existing near IR spectrometer from Flight 1 requires a larger environmental canister to house the instrument grouping. A SIPS heat pipe canister modified to extend its overall length by 20 inches and to provide MPM baseplate mounting is planned for this application. Descriptions of the deployed modules for Flight 2 can be found in Section 4.8.

4.1.4 Structures Analysis

Mass Properties - The following table summarizes the AMPS Flight 1 and 2 configuration weights.

	Launch Weight		Landed Weight	
	KGS	LBS	KGS	LBS
Flight 1				
Spacelab	6206	13682	6206	13682
Mission Dependent Equipment	2401	5293	1927	4248
Payload	6117	13485	4337	9562
Total Flight 1	14724	32460	12470	27492
Flight 2				
Spacelab	6206	13682	6206	13682
Mission Dependent Equipment	2772	6111	2298	5066
Payload	6212	13695	3922	8647
Total Flight 2	15190	33488	12426	27395

It can be noted from the table that there is a 4508 pound (2045 kg) margin for the Flight 1 return case. Flight 2 shows a similar margin. The center of gravity of each configuration, launch and landed, are within the constraints specified in Shuttle Orbiter Document JSC 07700, Vol. XIV. For Flight 1 a margin of approximately 29 inches (.74 metres) exists for the launch condition longitudinal CG, and a margin of approximately 40 inches (1.01 metres) is being shown for the landed condition. A cursory examination of the many possible variations in the payload configurations that exist for the aborted reentry and landing cases has been accomplished. These analyses show that the CG can be maintained for some of the more obvious cases. Appendix D contains the mass properties listing completed during the study. Launch and landed weights and CGs are presented for both Flights 1 and 2. A tabulation of the AMPS payload is given by separation into basic Spacelab, mission dependent, AMPS instruments, and AMPS Labcraft categories. The longitudinal CG plot is also presented in the appendix for Flights 1 and 2.

Dynamics and Vibroacoustics - The impact of the dynamics and vibroacoustics environments on the AMPS support structure design is discussed in Section 5.2.3. Structural test philosophy and recommendations for various testing including modal survey, vibro-acoustics, and static testing can be found in Section 5.2.4.

4.2 Thermal Control Subsystem

The Orbiter Spacelab/AMPS payload consists of numerous instruments and support equipment with diverse power and thermal requirements. Power dissipations vary from several watts to 5000 watts and component thermal environments range from the pressurized compartments to space. The AMPS thermal design matches the diversity of the requirements by use of Spacelab hardware and standard thermal control techniques to the maximum extent.

AMPS thermal design requirements include those imposed by the Spacelab (particularly on the fluid loop system) and those imposed by the instruments. Spacelab thermal requirements are obtained from the Spacelab accommodation handbook and instrument requirements from the Instrument Functional Requirement Document (IFRD). Development of the AMPS thermal design must consider specified vehicle attitudes with resultant environmental heating. The Spacelab thermal requirements are detailed in Section 4.2.1. Environmental heating requirements are discussed in Section 4.2.2. Key AMPS thermal design features are:

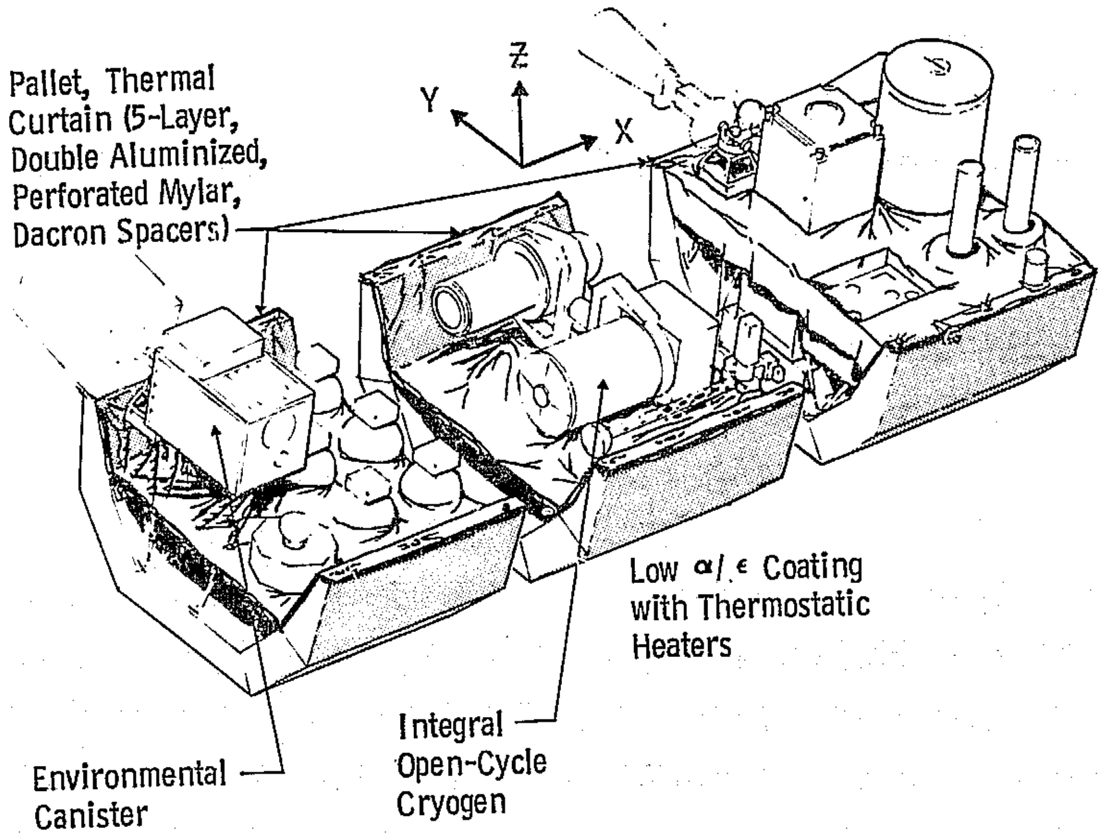
- Maximum use of an active thermal control loop and Spacelab hardware;
- A multilayer insulation (MLI) thermal curtain enclosing the pallets;
- A low α/ϵ external coating to cold-bias all components and use of thermostatic heaters for cold-case operation;
- Application of standard environmental canisters for instruments that require narrow temperature tolerances;
- Where required, recommendation of an open-cycle cryogen cooler integral with the instrument.

The AMPS thermal design approach has been developed considering all flights. The thermal performance has been verified analytically for the Flight 1 configuration. The Flight 1 thermal analysis shows that the AMPS baseline thermal control concepts meet component requirements and match the Orbiter/AMPS Spacelab payload thermal control capabilities. The thermal design concepts are general, and subsequent AMPS flights will perform similarly to Flight 1.

4.2.1 Thermal Control Requirements and Concepts

The AMPS baseline thermal control subsystem (TCS) is illustrated in Figure 4.2-1 for the Flight 1 configuration. The thermal design employs an active thermal control loop, pallet thermal curtain, cold-biased thermal design, environmental canister and open cycle cryogen.

Active Thermal Control Loop - The baseline AMPS/Spacelab payload



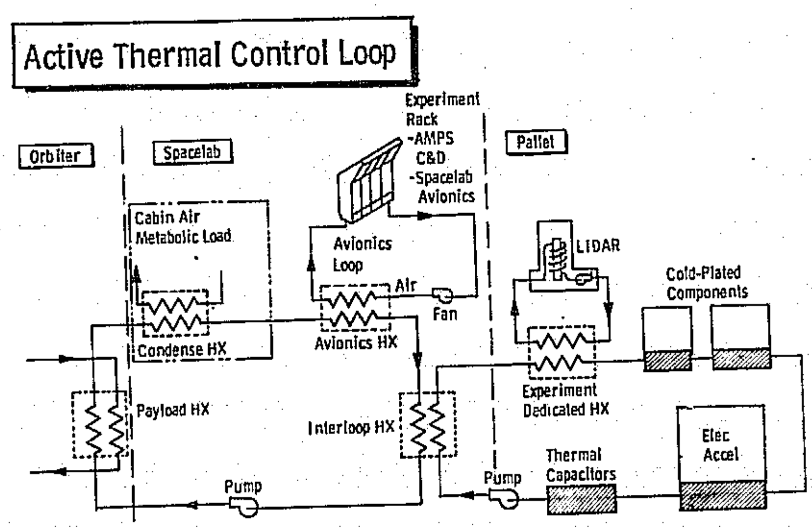


Figure 4.2-1 AMPS Thermal Design Summary

thermal control loop is illustrated in Figure 4.2-1. The Spacelab water loop thermally couples the pallet Freon-21 loop to the Orbiter payload heat exchanger that maintains the inlet water temperature to the Spacelab module below 45°F (7.2°C). The payload heat exchanger can remove an average of 8.5 kilowatts from the Spacelab coolant loop and pallet Freon temperatures vary between 75°F and 104°F (24 and 40°C). The Orbiter/Spacelab thermal control loop capabilities that are presented in the Spacelab Accommodation Handbook have been summarized in Table 4.2-1.

Table 4.2-1 Shuttle/Spacelab Thermal Control Capabilities

COMPONENT/SYSTEM	NOMINAL HEAT REJECTION WATTS	TEMPERATURE °C
Shuttle Payload HX		
Baseline with Add-On Radiator Kit		
- Prelaunch, Ascent, Entry and On-Orbit (Doors Closed)	1520	7.2 to Payload 37.8 from Payload
- On-Orbit (Doors Open)	8500	7.2 to Payload 40.0 from Payload
Spacelab		
- Avionics Loop Experiment Load	3000	22/40
- Cabin Air Experiment Load	1000	18/27
- Experiment Dedicated HX	4000	18/38
Pallet		
- Interloop HX	4850	24/40
- Eight Coldplates Provided with Basic Spacelab (Size -500x750 mm)	1000 (Each Coldplate)	
- Four Capacitor Elements Allocated (Size -500x700x25 mm)	0.270 kwh (Phase Change with Heneicosane)	40.2 Melt Point

Four thermal capacitors that utilize Heneicosane wax are allocated to the payload by Spacelab. A study has been completed (Section 5.3.3)

to determine the effects of thermal capacitors that are located at the outlet to the pallet Freon loop. The analysis shows that the thermal capacitors do not significantly affect coolant temperatures during operation of the high powered instruments and the capacitors have not been baselined. Investigation of instrument configurations has shown that it is often necessary to mount coldplates on the instruments. An alternative approach uses experiment-dedicated heat exchangers for concentrated heat loads such as the LIDAR. The AMPS payload requires components that are mounted in the Spacelab module on the experiment racks and on the Orbiter aft flight deck. The total heat loads for these components are below the required levels.

Pallet Thermal Curtain - Thermal analysis using the TRASYS and MITAS programs have shown that a thermal curtain enclosing the pallet is required for solar-inertial attitudes. Analysis results without the thermal curtain indicate that temperatures as high as 257°F (125°C) can be expected in cavities. A low α/ϵ multilayer insulation (MLI) structural curtain in conjunction with the pallet loop minimizes the temperature variations of the coldplated components.

Cold-Biased Thermal Design - A cold-biased semi-passive thermal design approach has been baselined for components that are mounted, fully or partially, outside the pallet thermal curtain. A low α/ϵ coating is used to maintain equipment at relatively low temperatures for hot conditions and thermostatic heaters are provided for cold case operations. The cold-biased thermal design approach is detailed in Section 5.3.2. The analyses show that this design approach meets thermal requirements.

Environmental Canister - The near-IR spectrometer and the spectrometer/photometer package used on Flight 1 contains complex optical elements. Comparison with similar instruments shows that the temperature fluctuations of the instrument case must be limited to approximately ± 9°F (± 5°C). Several instrument environmental canisters that provide relatively constant temperatures are currently being studied. Candidate approaches include the GSFC small-instrument pointing system (SIPS) heat pipe canister, an MSFC miniaturized pointing mount canister that uses Skylab Apollo Telescope Mount hardware, and a JSC AMPS instrument module system (AIMS) that uses coldplates coupled directly to the pallet loop. The Flight 1 AMPS configuration uses the MSFC liquid loop canister in conjunction with the mini-mount pointing platform. Heat rejection studies using the AMPS thermal math model have been completed and are discussed in Section 4.2.2.

Cryo-Cooled Instruments - The prime AMPS contractor is not directly responsible for the design of the cryo-cooled instruments. However, to facilitate development of a viable AMPS design, a preliminary study has been completed to select a cryogen instrument design concept. The limb scanner and the far-IR spectrometer require cryogenic temperatures. Radiators and thermoelectric devices are unacceptable because of their limited heat rejection capability. Open-cycle

coolers have been selected over mechanical refrigerators because of vibration, weight, and power penalties.

The limb scanner and the far-IR spectrometer themal design approach that has been baselined uses stored cryogens as an integral part of the instrument because this approach avoids the problems associated with the transfer of cryogens across pointing platform gimbals. Solid neon (20°K) is used to cool instrument walls and liquid helium (4°K) is used to cool the detectors.

4.2.2 Thermal Design Verification

The TRASYS program (environmental heating) and the MITAS program (subsystem temperatures) have been used to verify the performance of the AMPS thermal control subsystem. The analysis approach has been to construct a thermal math model that represents major elements of the TCS for the AMPS Flight 1 configuration. This configuration is typical of all AMPS flights and verification of the thermal performance of this configuration applies to subsequent flights. The MITAS thermal math model is described in Section 5.3.1. Preliminary AMPS thermal design studies have not identified potential problem areas during Orbiter launch and entry primarily because of the relatively short duration of these phases. Our analysis effort for the AMPS flights have emphasized orbital operations.

Thermal Environments - Verification of the AMPS thermal design requires the determination of the worst-case thermal attitudes. The parameters considered include environmental constants, beta angle (angle between the orbit plane and the sun) and vehicle attitude. Four cases have been selected to bound the AMPS flights thermal environments as shown in Figure 4.2-2. The TRASYS model used for the analysis is illustrated.

The normal attitude for the AMPS flights is with the Orbiter bay facing the earth (Z-LV attitude). This attitude facilitates studies of the earth's limb and is employed for approximately 80 percent of the flight. The beta angle range for a launch from the eastern test range (ETR) is from 0 to 80.5 degrees. A priori it is not clear whether the 0 or 80.5 degree beta angle in conjunction with Z-LV attitude is hotter and both cases have been investigated. Calibration measurements of the solar flux during the AMPS flights requires the Orbiter payload bay to face the sun for time periods of up to one orbit; thus, the third hot case vehicle attitude is the solar inertial case. Cold conditions occur for the AMPS payload with the payload bay facing away from the earth (-Z-LV) in conjunction with the 80.5 degree beta angle.

AMPS thermal performance analysis requires the coordination of the four environmental cases with other parameters to result in analysis cases that bound AMPS thermal performance.

Analysis Groundrules and Assumptions - Thermal analysis groundrules

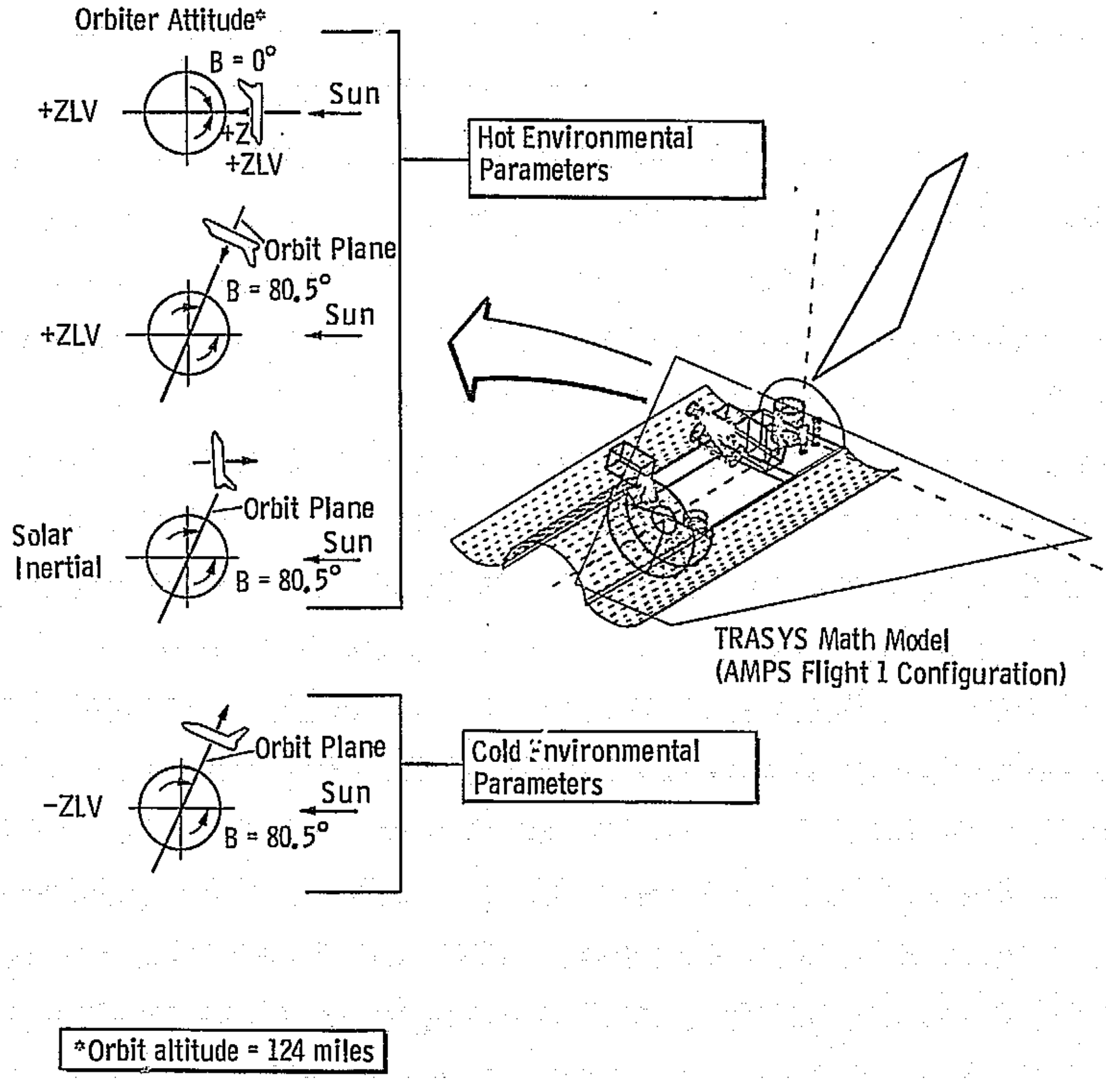


Figure 4.2-2 AMPS Worst Case Environmental Conditions

and assumptions for the four analysis cases are presented in Table 4.2-2. Included are the Orbiter bay door configuration, MLI blanket effective emittance, thermal coating properties, environmental parameters and electrical power. Power values are presented for the pallets, cold-biased components, mini-mount thermal canister, cryo-instruments and

Table 4.2-2 Thermal Analysis Groundrules, Guidelines and Assumptions

Item	Hot Case (ZLV)(1) B = 0 deg & B = 80.5 deg	Solar Inertial(2) Z Toward Sun B = 80.5 deg	Cold Case (-ZLV)(1) B = 80.5 deg
Orbiter Bay Doors Configuration	40°F (Fixed)	40°F (Fixed)	Float
Blanket Effective Emittances			
Thermal Curtain	0.04	0.04	0.04
Insulated Instruments	0.02	0.02	0.02
Cyro Instruments	0.01	0.01	0.01
Thermal Coatings			
AMPS External Surfaces	$\alpha = 0.25$, $\epsilon = 0.9$	$\alpha = 0.25$, $\epsilon = 0.9$	$\alpha = 0.25$, $\epsilon = 0.9$
Minimount Canister (Silverized Teflon)	$\alpha = 0.12$, $\epsilon = 0.76$	$\alpha = 0.12$, $\epsilon = 0.76$	$\alpha = 0.12$, $\epsilon = 0.76$
Environmental Parameters			
Orbit Altitude, mi	124.0	124.0	124.0
Solar Flux, Btu/hr-ft^2	457.0	457.0	401.0
Albedo Flux, Btu/hr-ft^2	192.0	192.0	72.0
Earth Flux, Btu/hr-ft^2	88.5	88.5	61.5
Pallet Power			
Pallets 1 & 2 (4 RAUs)	48 W	48 W	(0.8) (48) W
Pallet 3 (2 RAUs)	24 W	24 W	(0.8) (24) W
Cold-Biased Components			
Lidar Receiver	0 W	0 W	0 W
Pointing Platforms	100 W Each	(100 W Each)(3) 0 W	0 W
ESP	0 W	(148)(3) 0 W	0 W
Beam Diagnostics Package	0 W	(85)(3) 0 W	0 W
OBIPS	116 W	0 W	0 W
Gas Release Module	0 W	0 W	0 W
Minimount Environmental Canister	Radiator Inlet = 52°F (Maximum Flow)	Radiator Inlet = 52°F (Maximum Flow)	Radiator Inlet = 52°F (Minimum Flow)
Cryo Instruments			
Far-IR Spectrometer	-423°F, 10 W Electronics 46 W Telescope	-423°F, 10 W Electronics 46 W Telescope	-423°F, 10 W Electronics 46 W Telescope
Cryo Limb Scanner	-423°F, 40 W Electronics 35 W Telescope	-423°F, 40 W Electronics 35 W Telescope	-423°F, 40 W Electronics 35 W Telescope
Fluid Loop			
Status	On	On	On
Cabin Heater Exchanger	1406 W	1406 W	(0.8) 1406 W
Cabin Rack	1529 W	1529 W	(0.8) 1529 W
Water Loop Pump	53 W	53 W	(0.8) 53 W
Freon Loop Pump	274 W	274 W	(0.8) 274 W
Lidar Loop Pump	84 W	84 W	(0.8) 84 W
Lidar Transmitter	40 W Standby(4)	40 W Standby	(0.8) 40 W Standby
IECM	450 W	450 W	(0.8) 450 W
Power Supply	(0.2) (2159 W)(5)	0 W	0 W
Electron Accelerator	(0.8) (2159 W)(5)	0 W	0 W

(1) Quasi-Steady-State Analysis (Orbital Equilibrium).
(2) One Orbit Transient, Initial Conditions Hot-Steady-State, Beta = 0 deg.
(3) Internal/External Radiation Coupling Selection Analysis.
(4) 5000 W Peak During Operation; Lidar and Electron Accelerator Do Not Operate Together.
(5) Average Power; Peak Power of 5000 W Occurs during Dark Side of Orbit.

the fluid loop.

Analysis Results - Component temperatures/time histories presented in this section are for orbital equilibrium conditions except for the solar inertial attitude analyses that are for a one-orbit transient. Temperature requirements for Flight 1 instruments/components have been tabulated and compared to analysis results in Section 4.3.2 Note that component node identification numbers used in this section correspond to the thermal math model illustrated in Section 5.3-1.

The passive ESP and pallet thermal curtain temperature/time histories for hot conditions, as examples, are presented in Figure 4.2-3. The ESP is not powered in the stowed position and the temperature fluctuations shown are the result of variations in the orbital thermal environment. The internal ESP temperatures vary $\pm$ 5°F ($\pm$ 3°C), while the external skin changes 40°F (22°C). The thermal curtain has a low thermal mass and temperatures vary to a greater extent than the ESP. The average temperatures for the ESP and thermal curtain for the two beta angle extremes are similar. These results, and a review of temperatures of other components shows that the average component temperatures for the Z-LV attitude are a weak function of beta angle.

Hot/Cold Case - The hot-case temperature results for the fluid loop and the cold-case temperature results for the thermal curtain are presented in Figure 4.2-4. The temperature/time histories for the electron accelerator and the inlet to the payload heat exchanger show that the thermal environment of the two beta angle extremes has a negligible effect. The inlet to the payload heat exchanger and the electron accelerator are below the upper temperature limits. The thermal curtain temperatures show, for cold conditions, a maximum change during an orbit, of 70°F (39°C), with a low temperature of -210°F (-134°C).

Solar Inertial - Component temperature/time histories for the solar inertial attitude are presented in Figure 4.2-5, and include external surfaces, internal components/instruments and the fluid loop. Note that the initial conditions for these studies are hot steady-state (beta = 0.) and the analysis results are for a duration of one-orbit. Most equipment is powered down, as shown in Table 4.2-2, during the solar inertial condition.

The aft thermal curtain reaches 150°F (66°C), because of the congested aft pallet configuration. The midcurtain, electron accelerator, and LIDAR external temperatures peak at approximately 100°F (38°C). The mini-mount canister is stable at 50°F (10°C), primarily because of the canister coolant loop. The LIDAR and the OBIPS remain relatively constant and the ESP internal temperature increases 25°F (14°C). These temperatures are significantly below the 122°F (50°C) upper limit. Fluid loop temperatures and coldplated components drop in temperature during the solar inertial phase because these components were operating

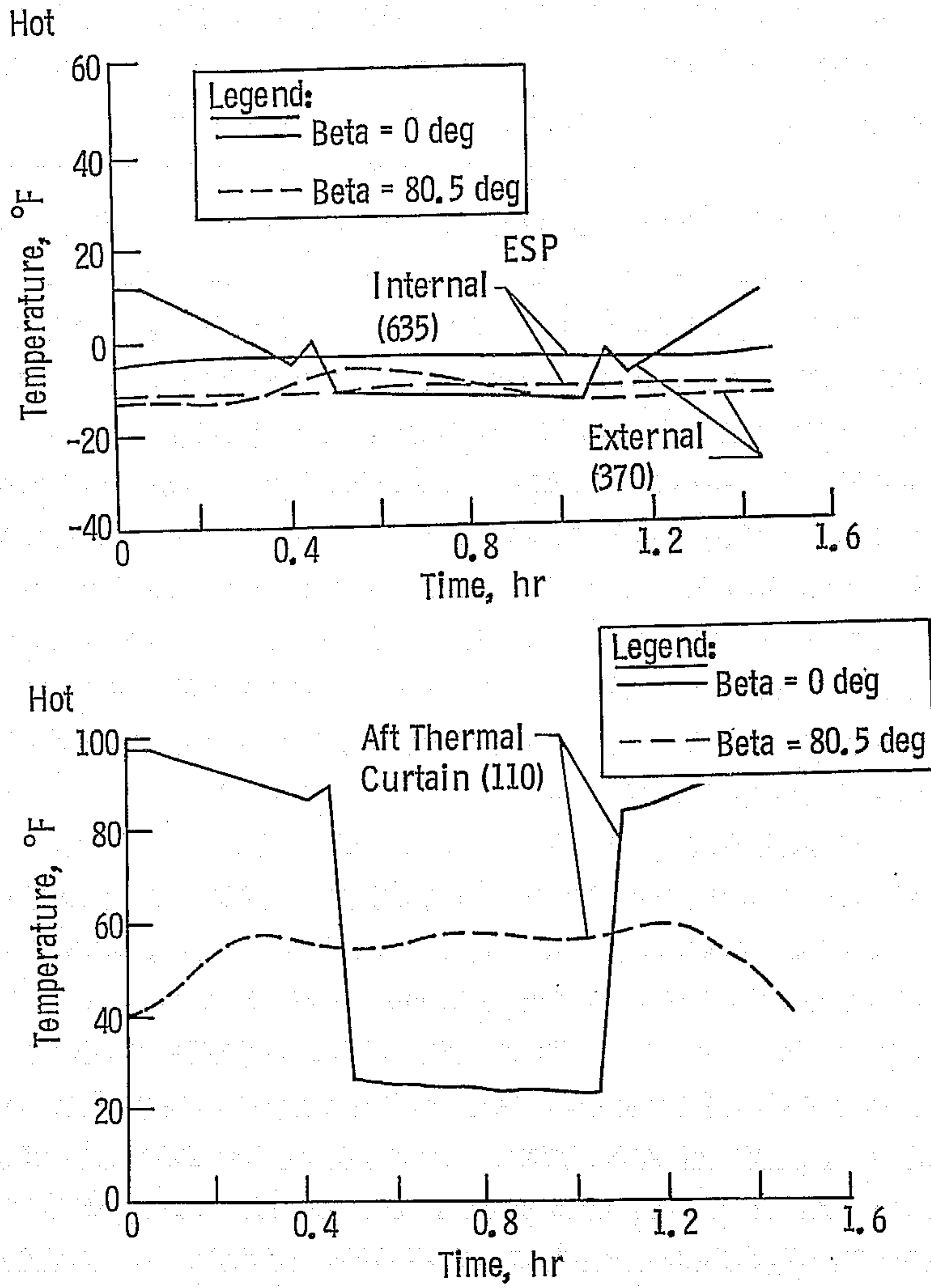


Figure 4.2-3 Hot Case Transient Thermal Analysis Results

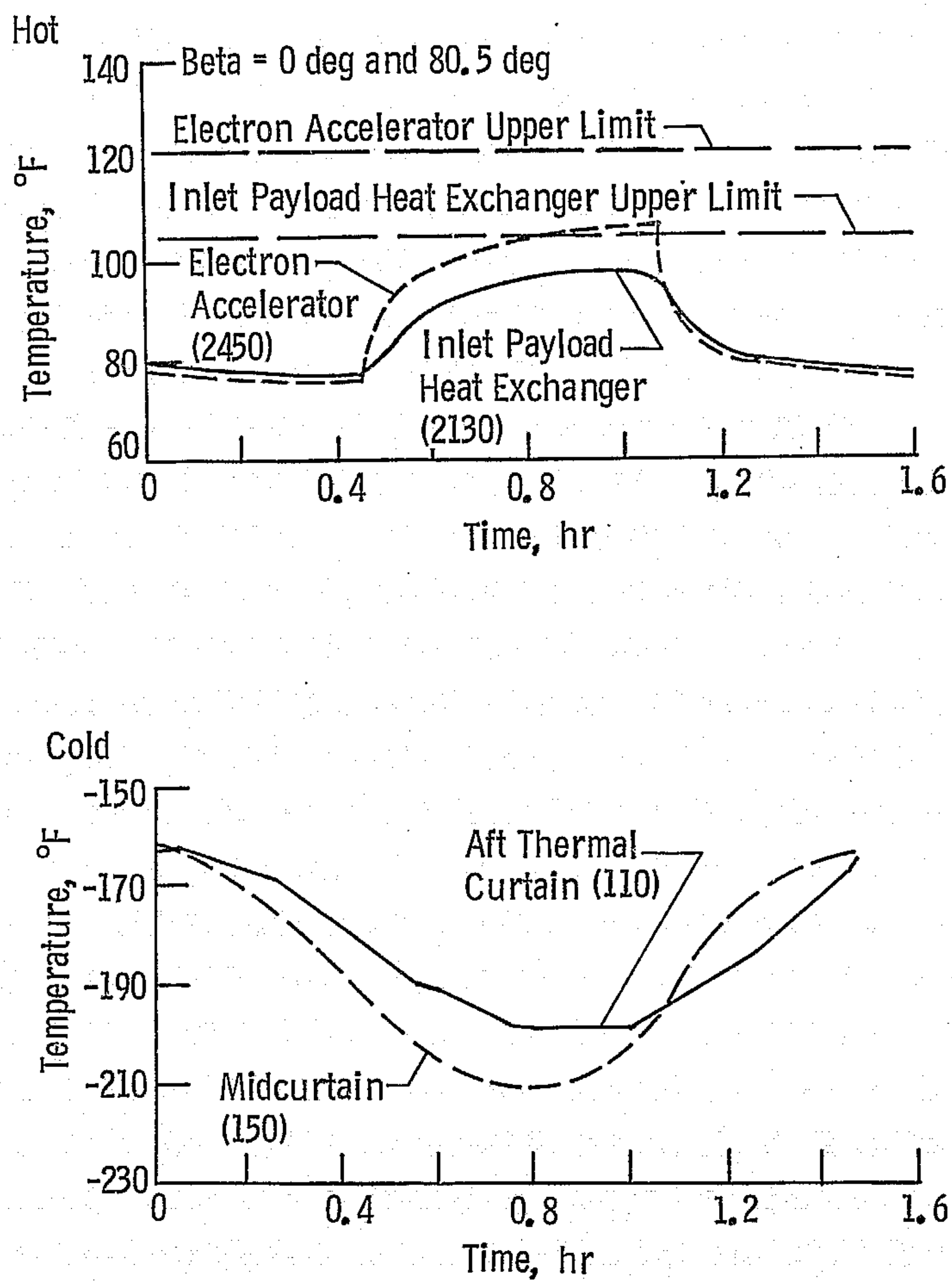


Figure 4.2-4 Hot Case/Cold Case Transient Thermal Analysis Results

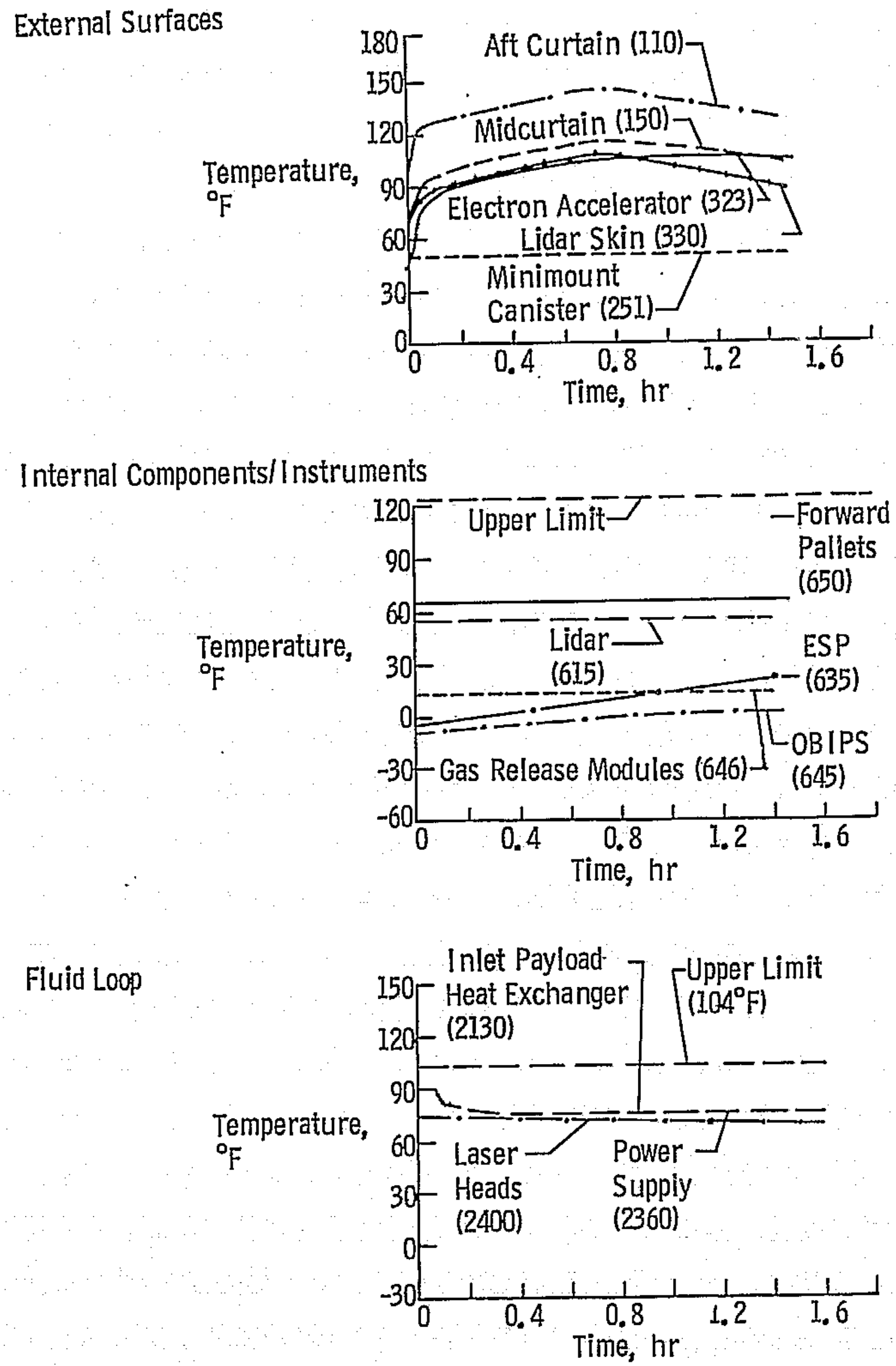


Figure 4.2-5 Solar Inertial Thermal Performance

at the beginning of the solar inertial phase and are powered down. All components meet requirements during the one-orbit solar inertial attitude operation.

Mini-mount Canister Heat Rejection - A study has been completed to determine the maximum heat rejection capability of the mini-mount thermal canister. The canister TCS schematic and analysis results are presented in Figure 4.2-6. Instrument heat is radiated to canister internal walls that are maintained below $52^{\circ}F$ ($11^{\circ}C$) by the liquid loop using a flow control valve and external radiator panels (external canister walls) as indicated.

The maximum canister heat rejection has been determined using the AMPS thermal model, by setting the canister radiator flow rate to the maximum value. Worst-case hot conditions and a 0.0 and 80.5 degree beta angle have been considered. The UV-VIS-NIR spectrometer/photometer (II-4 instrument) is housed within the thermal canister for Flight 1. The maximum power profile for the II-4 instrument is compared to the maximum canister heat rejection capability as shown. The worst-case power profile for the II-4 instrument should only be compared to the 0.0 degree beta angle maximum heat rejection because the II-4 instrument is used to study the earth's limb about sunrise/sunset. The 0.0 degree beta angle orbit has a day/night cycle whereas the 80.5 degree beta angle orbiter is always in the sun.

Comparison of the canister heat rejection with the II-4 instrument power shows that the mini-mount canister heat rejection capability meets requirements. Even for the full sun orbit (beta = 80.5°) the average canister heat rejection meets requirements of the II-4 instrument.

4.2.3 Thermal Performance Summary

The temperature requirements for AMPS Flight 1 components are compared with the analysis results in Table 4.2-3. The analysis shows that all components are below upper temperature limits and all cold-biased components are below temperature limits. Thermostatically controlled heaters are used to maintain the components at the lower temperature requirement.

Heater powers have been calculated for the cold-biased components and are presented in Table 4.2-4. A total of 288 watts is required. (mission average) and the total maximum heater power is 991 watts. The EPS analysis uses these heater powers and they are within the Orbiter capability.

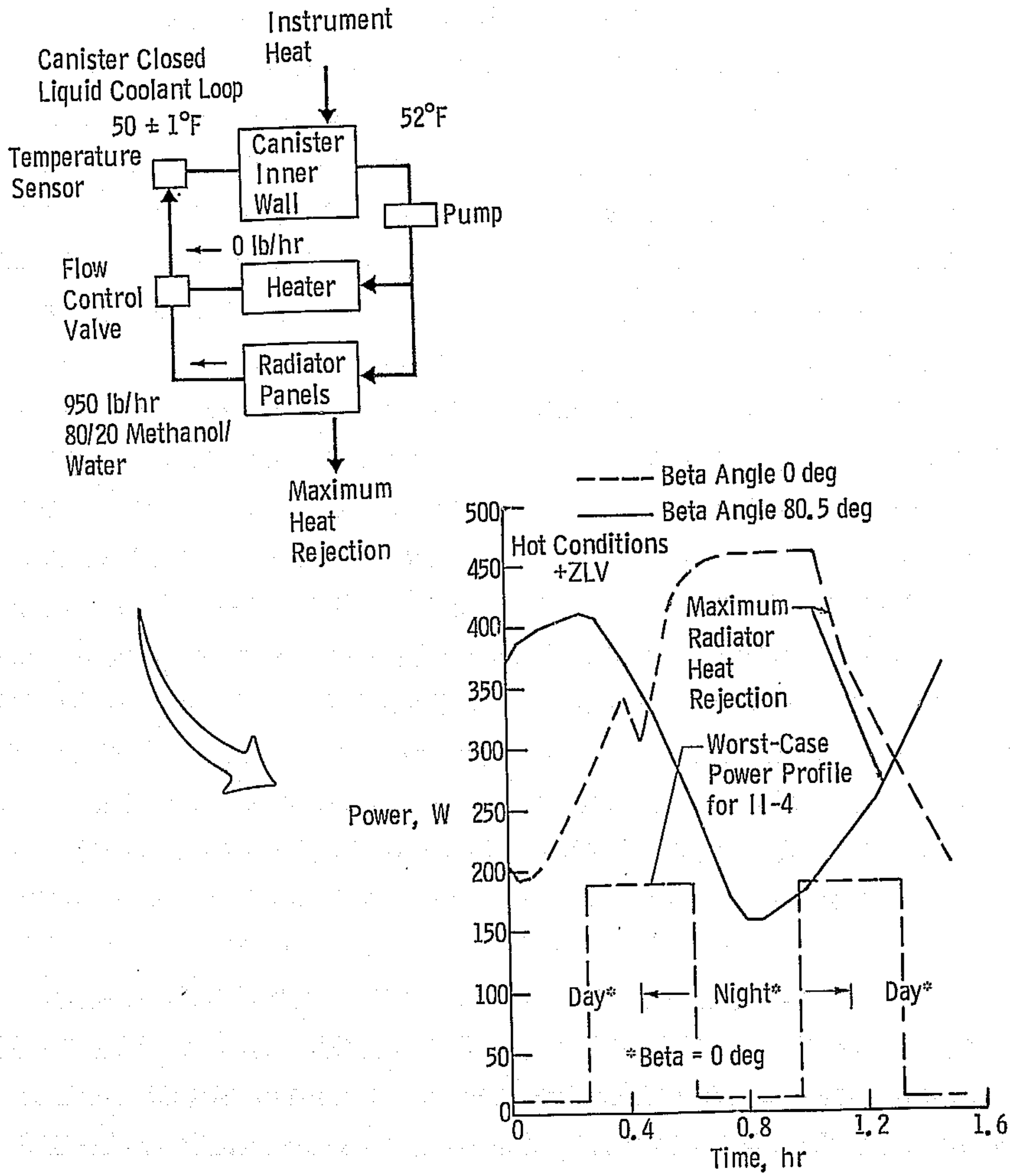


Figure 4.2-6 Mini-mount Canister Heat Rejection

Table 4.2-3 AMPS Flight 1 Thermal Performance Summary

Temperature, °F		
Component/Instrument	Requirements (2)	Analysis (5)
Lidar Receiver (1)	-22 to 122	-210 (6) to 55
Pointing Platforms (1)	-22 to 122	-134 (6) to 72
Minimount Canister Inner Walls	49 to 51	49 to 51
Minimount Canister Radiator	-100 to 100 (7)	-92 to 50
ESP (1)	0 (3) to 122	-181 (6) to 26
Beam Diagnostics (1)	0 (3) to 122	-203 (6) to 25
OBIPS (1)	-22 to 122	-219 (6) to 98
Gas Release Modules (1)	0 (3) to 122	-183 (6) to 15
Electron Accelerator (1)	-22 to 122	59 to 107
Laser Heads (1)	-22 to 122	59 to 117
Power Supply	-22 to 122	59 to 86
IECM	-22 to 122	65 to 85
Pallets	--	11 to 68
Aft Pallet Thermal Curtain	--	-198 to 144
Inlet to Payload Heat Exchanger	40 to 104 (4)	59 to 97

(1) Internal Instrument.
(2) IFRD
(3) Battery Storage Requirements, Heater Set Point, 0°F
(4) Space Shuttle Interface Control Document - Level II, JSC 12/17/75.
(5) Lowest/Highest Temperature Achieved during the Four Analysis Cases for Transient Conditions.
(6) Controlled to Lower Limit with Thermostatic Heater.
(7) Skylab ATM Requirement.

Table 4.2-4 AMPS Flight 1 Heater Power

Heater Power, W		
Heater Location	Mission Average (1)	Maximum (2)
Lidar Receiver	8	20
Pointing Platforms (3)	24	186
Minimount Canister	138	345
ESP	60	230
Beam Diagnostics	34	140
OBIPS	12	50
Gas Release Modules	12	20
Total	288	991

(1) Based on 80% +ZLV Hot (Beta = 0 deg) and 20% +ZLV Cold (Beta = 80.5 deg), Components Off.
(2) Cold Conditions, Components Off.

4.3 Electrical Power and Distribution System (EPDS)

The Shuttle Orbiter fuel cells provide the primary source of power for the AMPS electrical loads. The fuel cell power is switched through the Spacelab distribution system and is available to the AMPS loads at the Electrical Power Distribution Boxes (EPDB) on each pallet. The ac power requirements for AMPS loads are satisfied by the use of the Spacelab 400 Hz inverter which is mounted in the pressurized module. The ac power is also switched through the Spacelab distribution system and is available to AMPS loads at the EPDB interfaces. The fuel cells provide the AMPS payload an average power of 3400 watts at a nominal 28 volts with peak energy of 7400 watts available for 15 minutes once every 3 hours. The energy available to the AMPS payload is 369 kilowatt-hours.

4.3.1 Electrical Power and Distribution Requirements and Concepts

The EPDS consists of: Electrical Distribution Units (EDUs) for load switching and circuit protection; Pyro Initiator Units for instrument release; Power Supplies for deployed instrument power and peak load requirements; Pulsed Power Supply for high voltage; high energy and interconnecting cabling for distribution of power, commands and telemetry. Figure 4.3-1 and Figure 4.3-2 show the EPDS configuration for Flights 1 and 2 respectively. Figure 4.3-3 is a summary of the EPDS hardware and gives the design/development status and key features of each component.

Science instrument power requirements used in the design of the EPDS were obtained from instrument IFRDs. Figures 4.3-4 and 4.3-5 are a listing of these requirements and show a proposed location for each instrument. Inasmuch as possible, Spacelab EPDS hardware has been utilized to form the basic power distribution system with AMPS dedicated hardware being added to provide switch and circuit protection for AMPS electrical loads.

Power Distribution - The power requirements of the AMPS payload are divided into two categories:

(1) Pallet Mounted - Power for mounted pallet loads is obtained from the Orbiter fuel cells via the Spacelab Power Bus using AMPS EDUs and interconnecting cabling. Power from the Spacelab bus is switched in the EDUs and distributed to the pallet loads. Switched power distributed to the deployable instruments provides heater power prior to release from the pallets.

(2) Deployable Instruments - Instruments that are deployed either as a free-flying subsatellite or as a recoverable instrument package receive power from a storage battery internal to the instrument package. Batteries are activated prior to installation and remain in a ready state until power is required.

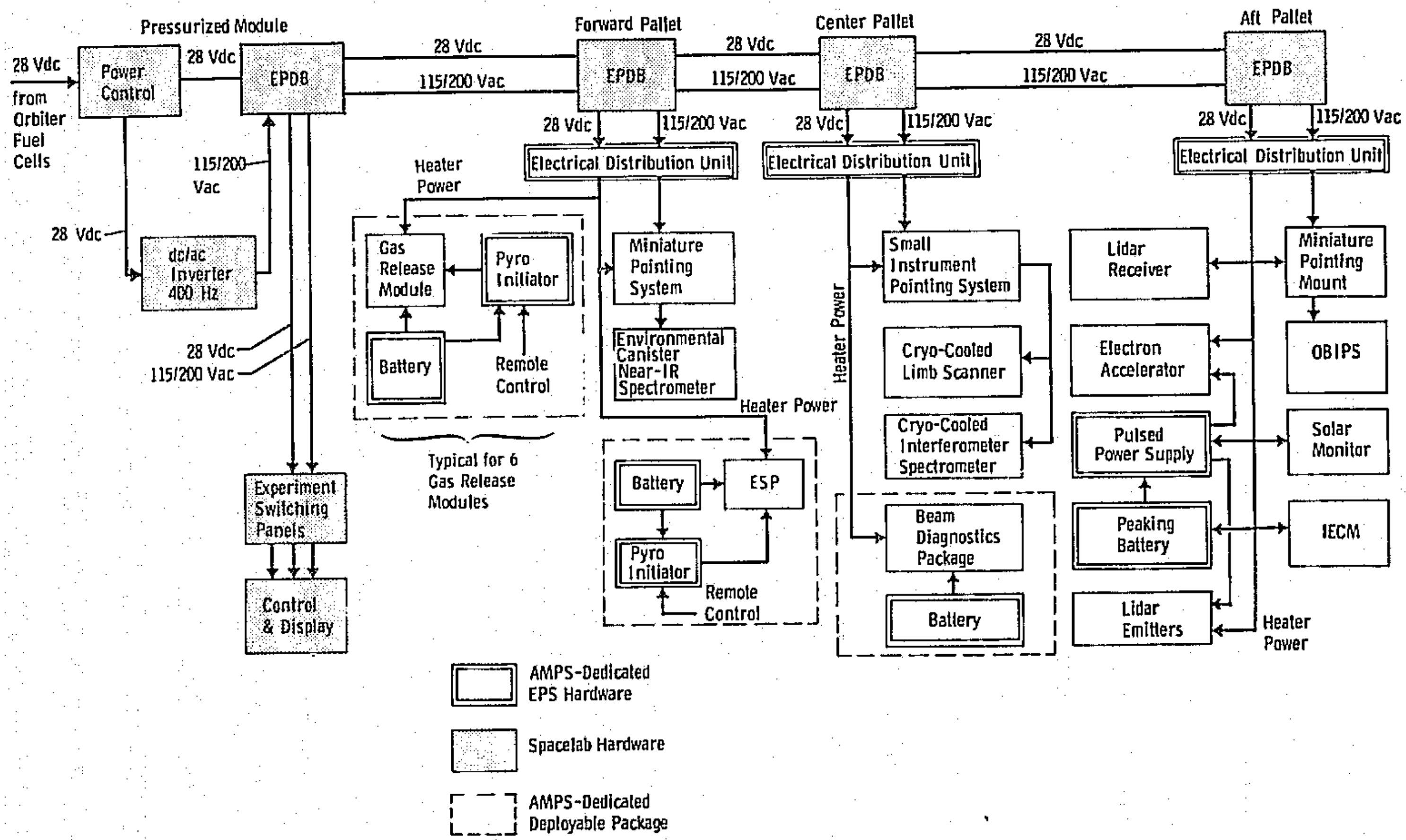


Figure 4.3-1 EPDS Block Diagram, Flight 1

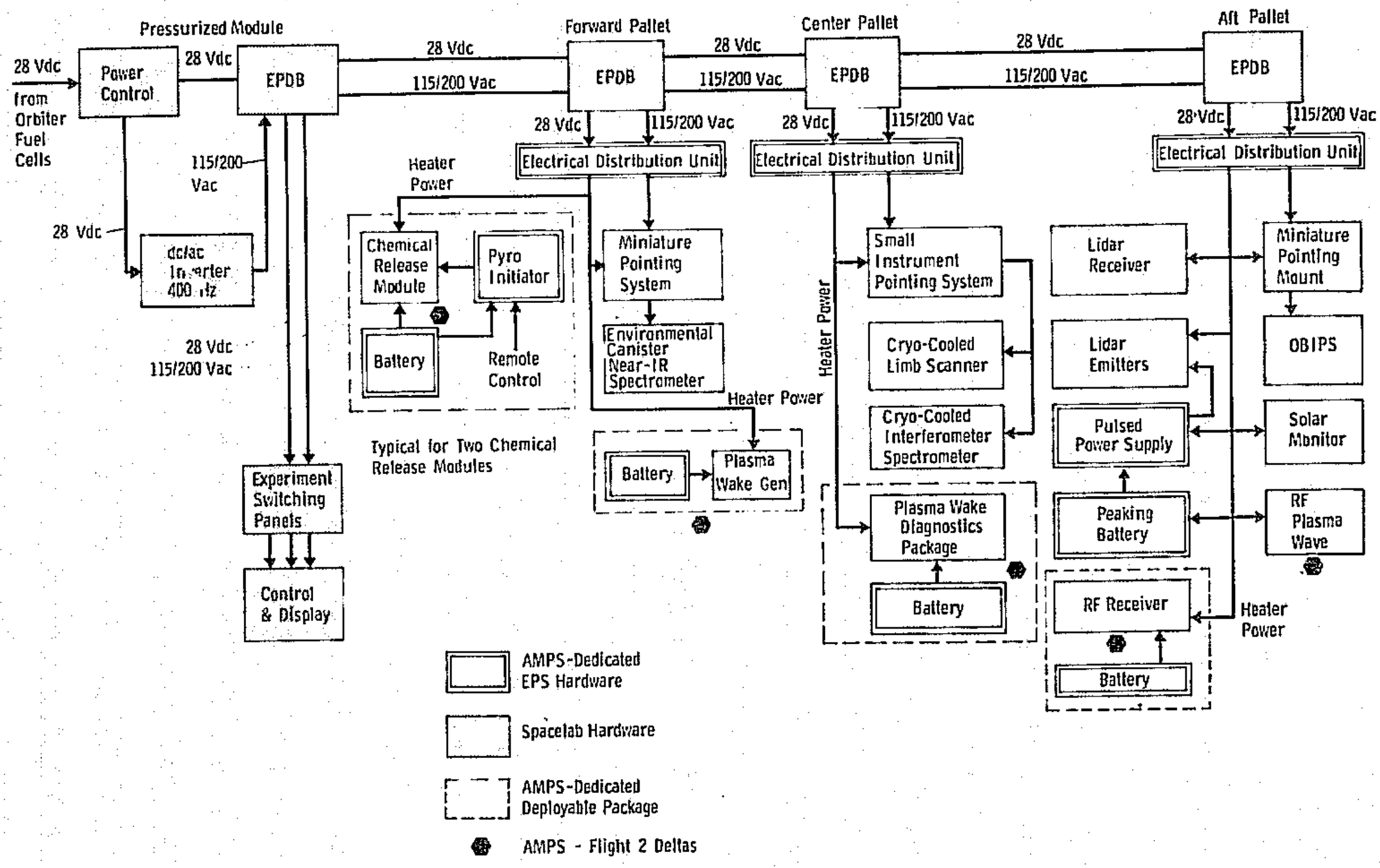


Figure 4.3-2 EPDS Block Diagram, Flight 2

A. FLIGHT 1 EPDS EQUIPMENT

NOMENCLATURE	QTY	VENDOR	PREVIOUS USAGE	MODIFICATION DESCRIPTION
Inverter, 400 Hz	1	Spacelab Furnished	Spacelab	None
Electrical Distribution Unit	3	MMC Build	Viking Lander	Modified Viking
Peaking Battery Package	1	MMC Build	New Design	None
- 160 AH Battery	1	Eagle Picher	Titan	None
- Battery Charger	1	Engineered Magnetics	Rolland Missile	Modified Output
Pulsed Power Supply	1	Hughes, TRW	New Design	None
- LIDAR	1	Hughes, TRW	New Design	None
- Electron Accelerator	1	Hughes, TRW	New Design	None
Pyro Initiator - Gas Release	12	MMC Build	Shuttle	None
Pyro Initiator - ESP	1	MMC Build	Shuttle	None
Power Supply - Gas Release	6	MMC Build	New Design	None
- 1.7 AH Battery	6	Eagle Picher	Titan	None
Power Supply - ESP	1	MMC Build	New Design	None
- 105 AH Battery	1	Eagle Picher	Titan	None
Power Supply - Beam Diagnostics	1	MMC Build	New Design	None
- 160 AH Battery	1	Eagle Picher	Titan	None
Cabling		MMC Build	Similar to Skylab MDA	New Design

B. FLIGHT 2 EPDS EQUIPMENT

NOMENCLATURE	QTY	VENDOR	PREVIOUS USAGE	MODIFICATION DESCRIPTION
Inverter, 400 Hz	1	Spacelab Furnished	Spacelab	None
Electrical Distribution Unit	3	MMC Build	Viking Lander	Modified Viking
Peaking Battery Package	1	MMC Build	New Design	None
- 160 AH Battery	1	Eagle Picher	Titan	None
- Battery Charger	1	Engineered Magnetics	Rolland Missile	Modified Output
Pulsed Power Supply	1	Hughes, TRW	New Design	None
- LIDAR	1	Hughes, TRW	New Design	None
Pyro Initiator - Chem Release	4	MMC Build	Shuttle	None
Pyro Initiator - RF Receiver	1	MMC Build	Shuttle	None
Power Supply - Chem Release	2	MMC Build	New Design	None
- 1.7 AH Battery	2	Eagle Picher	Titan	None
Power Supply - Wake Gen.	1	MMC Build	New Design	None
- 160 AH Battery	1	Eagle Picher	Titan	None
Power Supply - Wake Diagnostics	1	MMC Build	New Design	None
- 65 AH Battery	1	Eagle Picher	Titan	None
Power Supply - RF Receiver	1	MMC Build	New Design	None
- 16.8 AH Battery	1	Eagle Picher	Titan	None
Cabling		MMC Build	Similar to Skylab MDA	New Design

C. Electrical Distribution Unit

- Establishes Redundant Busses
- Switches Input Power to Either Bus
- Provides Redundant Power to Each Instrument
- Provides Circuit Protection For All Circuits
- Redundant Digital Interface Circuits Process Switching Commands From RAUs.

D. Battery

- AgZn Primary Batteries
- Existing Designs
- Flight Qualified
- Shelf Life - 2 to 5 Yrs (Dry)
- Storage Life - 15 to 30 Days (Wet)
- Provides Energy for Deployable Packages
- Provides Energy for Peaking Package
- Rechargeable for Peaking Package

E. Battery Charger

- Existing Design
- Flight Qualified (Prior to AMPS Usage)
- 30 Amperes Output
- 22 to 35 Volt Input
- 24 to 36 Volt Output
- Automatic Charge Control
- Ground or Onboard Control Override
- Recharges Peaking Battery

F. Pyro Initiator

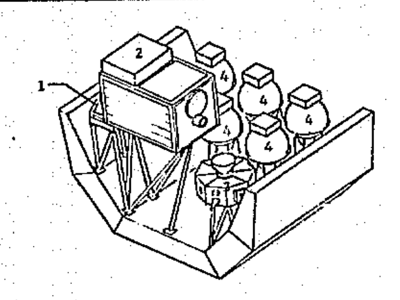
- Existing Design
- Flight Qualified (Prior to AMPS Usage)
- Provides Redundant Fire Signals
- Provide Safe/Arm Function
- Releases Components From Pallets
- Releases Gas and Chemicals in Deployed Packages

G. Cabling

- Provides Distribution of Power, Signal and Commands
- Category Separation
- Existing Design Techniques
- Simplicity of Design
- Universal Harness Design Where Possible
- Flight Proven, Qualified Connectors
- Interconnect at Pallet Interface

Figure 4.3-3 EPDS Design and Configuration Summary

FLIGHT I POWER INTERFACES

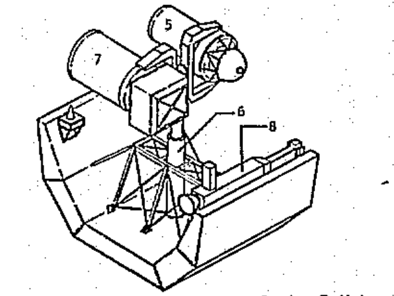


Forward Pallet

1 MINI POINTING MOUNT
2 NEAR IR SPECTROMETER
3 ESP
4 GAS RELEASE MODULES

	POWER REQUIREMENT	
	AVERAGE	STANDBY
II-9		
NEAR IR SPECTROMETER	196 W	10 W
MINI POINTING MOUNT	200 W	115 W
MINI MOUNT PUMP	115 W	115 W
THERMAL	162 W	162 W
I-21		
GAS RELEASE SYSTEM(1)	2 W/UNIT	-
THERMAL(2)	12 W	-
ESP(1)		
III-16 ION MASS & DIST ANALYZER.	10 W	7 W
III-18 PLANAR RPA	5 W	0 W
III-22 LANGMUIR PROBE	5 W	0 W
III-23 NEUT MASS SPECT	40 W	1 W
III-25 EMI DIAGNOSTICS	10 W	7 W
THERMAL(2)	60 W	60 W

(1) ALL POWER SELF-CONTAINED - NO LOAD ON SPACELAB BUS
(2) CONTINUOUS PRIOR TO MODULE DEPLOYMENT

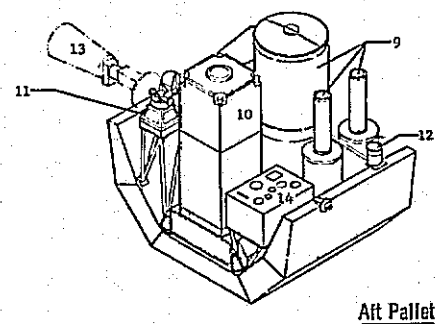


Center Pallet

5 CRYO COOLED LIMB SCANNER
6 SIPS
7 CRYO COOLED INTERFEROMETER SPECTROMETER
8 BEAM DIAGNOSTICS

	POWER REQUIREMENT	
	AVERAGE	STANDBY
II-7		
CRYO COOLED LIMB SCANNER	75 W	15 W
SIPS	268 W	59 W
THERMAL		
II-10		
CRYO COOLED IR SPECTROMETER	56 W	0.5 W
BEAM DIAGNOSTICS(1)		
II-3 OBIPS		
III-2 VECTOR MAGNETOMETER	179 W	12 W
III-4 LEVEL II BEAM DIAGNOSTICS		
THERMAL(2)	34 W	34 W

(1) ALL POWER SELF-CONTAINED - NO LOAD ON SPACELAB BUS
(2) CONTINUOUS PRIOR TO MODULE DEPLOYMENT



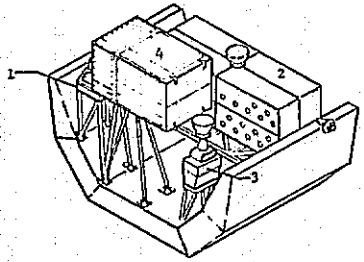
Aft Pallet

9 LASER SOUNDER (RECEIVER AND TRANSMITTERS)
10 ELECTRON ACCELERATOR
11 MINI POINTING MOUNT
12 SOLAR FLUX CAL ARRAY
13 OBIPS
14 IECH

	POWER REQUIREMENT	
	AVERAGE	STANDBY
I-1		
LASER SOUNDER	3600 W	40 W
LIDAR PUMP	84 W	84 W
THERMAL	8 W	8 W
I-9		
ELECTRON ACCELERATOR	2500 W	200 W
II-3		
OBIPS	116 W	7 W
MINI POINTING MOUNT	200 W	115 W
THERMAL	20 W	20 W
IV-1		
SOLAR FLUX CAL ARRAY	8 W	-
IECH	400 W	400 W

Figure 4.3-4 Flight 1 Electrical Power Requirements

FLIGHT 2 POWER INTERFACES

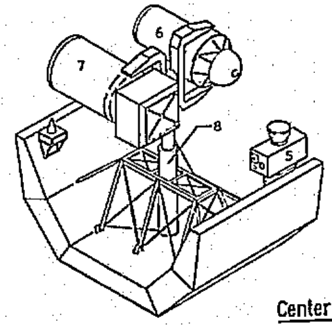


Forward Pallet

1 MINI POINTING MOUNT
2 CHEMICAL RELEASE SYSTEM
3 PLASMA WAKE GENERATOR
4 UV-VIS-NIR SPECTROMETER/PHOTOMETER

	POWER REQUIREMENT	
	AVERAGE	STANDBY
I-21		
CHEMICAL RELEASE SYSTEM(1)	2 W/UNIT	0.5 W
THERMAL(2)	12 W	12 W
II-4		
UV-VIS-NIR SPECTROMETER/ PHOTOMETER	60 W	0 W
MINI POINTING MOUNT	200 W	115 W
PUMP	115 W	115 W
THERMAL	146 W	146 W
PLASMA WAKE GEN(1)		
III-17		
DEPLOYABLE TEST BODY	~10 W	

(1) ALL POWER SELF-CONTAINED - NO LOAD ON SPACELAB BUS
(2) CONTINUOUS PRIOR TO MODULE RELEASE

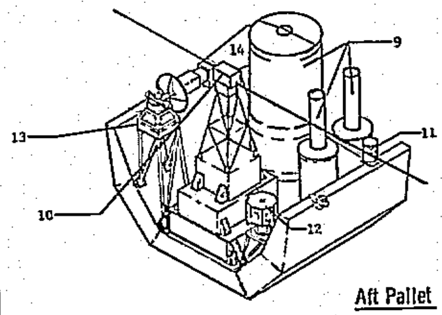


Center Pallet

5 PLASMA WAKE DIAGNOSTICS
6 CRYO COOLED LIMB SCANNER
7 CRYO COOLED INTERFEROMETER SPECTROMETER
8 SIPS

	POWER REQUIREMENT	
	AVERAGE	STANDBY
II-10		
CRYO-COOLED IR SPECTROMETER	56 W	0.5 W
II-7		
CRYO-COOLED LIMB SCANNER	75 W	15 W
SIPS	268 W	59 W
THERMAL		
PLASMA WAKE DIAGNOSTICS(1)		
III-16 ION MASS & DIST ANALYZER	10 W	7 W
III-18 PLANAR RPA	5 W	0 W
III-2 VECTOR MAGNETOMETER	40 W	5 W
III-22 LANGMUIR PROBE	5 W	0 W
III-23 NEUTRAL MASS SPECTROMETER.	40 W	1 W
THERMAL(2)	60 W	60 W

(1) ALL POWER SELF-CONTAINED - NO LOAD ON SPACELAB BUS
(2) CONTINUOUS PRIOR TO MODULE DEPLOYMENT



Aft Pallet

9 LASER SOUNDER (RECEIVER AND TRANSMITTERS)
10 RF PLASMA WAVE GENERATOR
11 SOLAR FLUX CAL ARRAY
12 RF RECEIVER PACKAGE
13 MINI POINTING MOUNT
14 OBIPS

	POWER REQUIREMENT	
	AVERAGE	STANDBY
I-12		
RF PLASMA WAVE PACKAGE	220 W	20 W
II-3		
OBIPS	116 W	7 W
MINI POINTING MOUNT	200 W	115 W
THERMAL	12 W	12 W
I-1		
LASER SOUNDER	3600 W	40 W
LIDAR PUMP	84 W	84 W
THERMAL	8 W	8 W
RF RECEIVER PACKAGE		
I-12 RF PLASMA WAVE PKG	220 W	20 W
III-2 VECTOR MAGNETOMETER	40 W	5 W
THERMAL	34 W	34 W
IV-1		
SOLAR FLUX CAL ARRAY	8 W	-

Figure 4.3-5 Flight 2 Electrical Power Requirements

Power switching is controlled either through an RF link to the deployed package command receiver or by an internal command from the using instrument.

Electrical Distribution Units - The EPDS contains three EDUs, one located on each pallet. The EDU provides circuit protection and load control for each AMPS load. The EDU switching commands come from Remote Acquisition Units (RAUs) mounted on each Spacelab pallet. Processing of RAU commands is accomplished through use of digital signal decoding interfaces operating the magnetic latching relays. Decoding interfaces are redundant and either interface will perform the required switching. EDUs provide bus redundancy, bus isolation, and overload protection for each output circuit and are capable of transferring both 28 vdc and 400 Hz, 115 vac. EDUs provide single, series redundant, and parallel redundant relay switching. Typical switching functions include:

- Operational power and heater power for instruments operating on the Spacelab pallets;
- Pyrotechnic release power for deployable instruments;
- Heater power for deployable instruments while pallet mounted.

Pyro Initiator Units - Actuation of explosive release devices is controlled by Pyro Initiator Units (PIUs). The PIU takes voltage from the EDUs and switches it to the release device when a command is received from a RAU or command receiver. A Pyro Actuation Unit provides a safe/arm function and built-in test points to determine the status of the release device. The PIUs are internally redundant for each pyro event to insure proper firing.

Power Supplies - Instrument packages which are removed from their pallet mountings during the operational sequence contain internal power supplies. The power supply design selected as the result of the trade study in Section 5.4.1 uses silver-zinc primary batteries. The batteries are not recharged on orbit and are therefore sized to meet the instrument package energy requirements with a positive power margin, including allowance for power increases and contingency operation. The battery design selected is a flight qualified design available off-the-shelf in a range of sizes from 1 to 200-ampere hours. Shelf storage life is 2 to 5 years dry and 15 to 30 days wet. Ninety to 100% of the nominal capacity is available after 15 days of wet stand.

A 160 ampere-hour source power supply is included in the peaking battery package. This application of a flight qualified, primary silver-zinc battery requires a limited number of charge/discharge cycles to support high power usage experiments. A flight qualified battery charger and the distribution and control logic complete the design of the peaking battery package.

Pulsed Power Supply - The Pulsed Power Supply is required to provide high voltage, high energy dc pulsed power for instrument operation. The pulsed power supply design is a dual tier power processing design with individual power processing units for each instrument as shown in Figure 4.3-6. The design shown includes a 1.3 farad capacitor bank to provide 100 kilojoules of energy to the Electron Accelerator on Flight 1. The 2 kilojoule requirement of the LIDAR Emitter for Flights 1 and 2 is also satisfied using the common capacitor storage bank. Subsequent flights include instruments with high voltage, high energy pulses exceeding the Flights 1 and 2 requirements by orders of magnitude. The exact energy levels are not known at the present time because pulse durations are either not defined or depend on system operating modes.

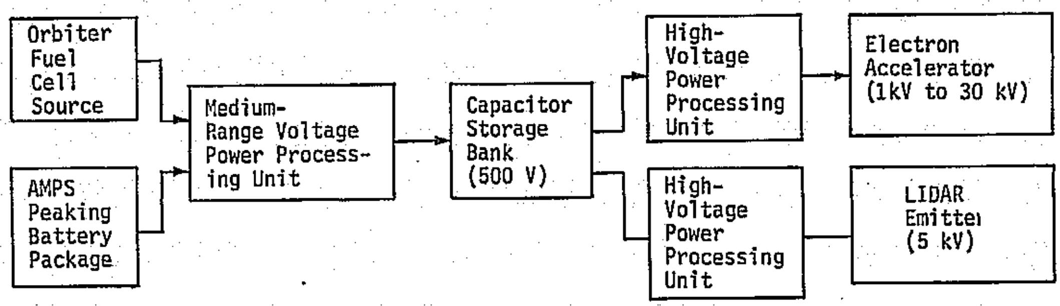


Figure 4.3-6 DC Pulsed Power Supply Configuration

Table 4.3-1 is a summary of the voltage and current levels for each AMPS high voltage instrument. Pulse durations are given where they are available. It can be seen that even very short duration pulses at the maximum voltage and current levels will result in energy requirements in the megajoule range. Capacitor storage banks to provide several megajoules of energy may not be acceptable because of weight and volume constraints. At 100 joules per pound, a 1 megajoule capacitor bank would weight 10,000 pounds. Therefore, as the requirements for these instruments are defined, additional studies will be required to identify design techniques that will satisfy the instrument requirements within the weight and volume constraints of a Spacelab payload. Of special interest in such a study would be the use of energy storage devices using rotational kinetic energy such as flywheels or homopolar generators. An additional technique of interest

is magnetic field energy storage using super conducting coils cryogenically cooled to temperatures of approximately 2°K. The use of batteries and dedicated fuel cells for energy storage should also be considered. The batteries and fuel cells may be required to provide the energy and power levels required to recharge the pulsed power supply selected if the pulse repetition rates exceed the Orbiter fuel cell output capability.

Table 4.3-1 High Voltage Power Supply Requirements

Instrument	Avg. S/C Power	High Voltage Characteristics		
		Voltage	Current	Pulse
I-1 LIDAR	1-5 KW	5 KV	400 A	1 msec
I-5 MPD Arc	5 KW	100 V to 500 V	1 KA to 250 KA	20 to 200 msec
I-6 H.V. Plasma Guns	5 KW	1 KV to 10 KV	10 KA to 250 KA	System Dependent
I-7 SEPAC				
a) Electron Gun	1.2 KW	0.1 KV to 40 KV	0.1 A to 10 A	Variable Modes
b) Ion Source	1 KW	1 KV to 20 KV	0.2 A to 1 A	System Dependent
c) MPD Arc	2 KW	150 V to 250 V	10 to 360 KA	1-3 MS
I-9 Electron Accelerator	5 KW	1 KV to 30 KV	0-7 A	System Dependent

Interconnecting Cabling - Interconnecting cabling provides for the transfer of power, signals, and commands. The AMPS cabling will provide the following: 1) minimum loss of power, signal, and commands due to any mission environment; 2) sensitive circuit (EMI) protection by means of physical cable separation and/or circuit shielding; 3) construction utilizing standard hardware designed for space operation; 4) use of commercial hardware where possible; 5) simplicity of design to assure minimum construction cost and ease of maintenance; and 6) installation techniques that eliminate mechanical interferences. Special emphasis will be directed towards pyro circuits and range safety requirements will be strictly adhered to.

The cabling design concept provides for disconnect points at inter-pallet separations and hard mounted instrument interfaces. Where possible, disconnects are accomplished with connectors but other techniques will be employed when connectors cannot be used. Each self-contained portion of cabling will be built separately to provide for ease of manufacturing, installation and maintenance. The cabling will be built

either on the AMPS pallets and support structure or on a three dimensional fixture to assure correct positioning and proper fit. AMPS cabling design will also include close coordination with and inputs to each instrument manufacturer to obtain a maximum amount of standardization between instruments where standardization allows construction of universal cabling without affecting instrument operation. Areas that are considered are: 1) connector locations with respect to instrument mounting, 2) connector types used and the purpose for each type, 3) the routing of wires to a connector and the position of each wire in a connector.

4.3.2 Trade Studies and Analyses

EPDS trade studies and analyses were completed to optimize the design configuration at the minimum program cost. This section presents a brief summary of the study results; complete studies are found in Section 5.4 and Appendix E.

Electrical Energy Management - A power profile analysis of the Flight 1 mission was completed using the mission timelines generated during the study and discussed in Section 3.2. The power requirements used for the instruments are those specified in instrument IFRDs. Groundrules and assumptions used in the course of the analyses were:

a. Night cycle duration per orbit = 37 minutes
   Day cycle duration per orbit = 53 minutes;

b. Instrument operation when not indicated as a full day or night cycle was estimated and an average power was calculated for use in the analysis;

c. Power for the experiment computer and I/O unit is part of the Spacelab power allocation.

A power profile for the first day of flight is shown in Figure 4.3-7. The detailed analysis and power profiles for days 2 through 6 are in Appendix E. A summary of the total Flight 1 energy requirements is shown in Table 4.3-2. The study results show the energy usage to be 378 kwh. This value, although 2.5% greater than the allocated 369 kwh, is one that will be easily manageable once actual energy requirements for science instruments are established.

The power analysis for Flight 2 was not performed since the majority of the science instruments were the same as Flight 1. This indicated that the energy used on Flight 2 would approximate that of Flight 1 and would also be manageable with establishment of actual requirements.

RF Versus Hardwire Trade Study - A study was performed to select the design configuration for providing power, data and commands to the

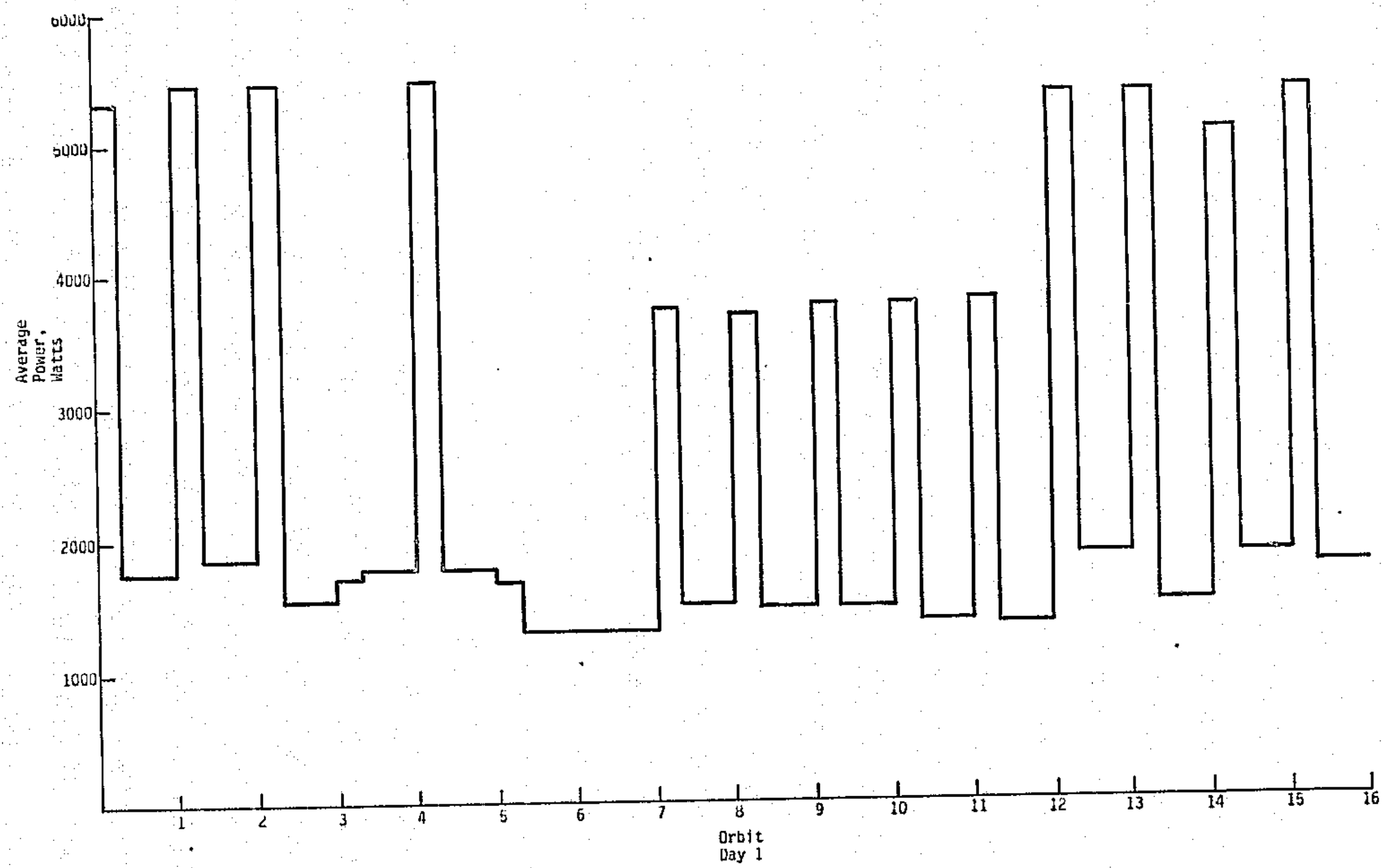


Figure 4.3-7 Mission Day 1 Power Profile

Table 4.3-2 Flight 1 Energy Usage Summary

	DAY 1		DAY 2		DAY 3		DAY 4		DAY 5		DAY 6		MISSION TOTAL (WH)
	ORBITS 1-8	ORBITS 9-16	ORBITS 1-8	ORBITS 9-16	ORBITS 1-8	ORBITS 9-16	ORBITS 1-8	ORBITS 9-16	ORBITS 1-8	ORBITS 9-16	ORBITS 1-8	ORBITS 9-16	
I-1 Laser Sounder	9,045	9,020	9,045	6,825	9,045	6,825	13,650	15,845	13,650	15,845	11,300	6,755	126,850
II-7 Cryo Cooled Limb Scanner SIPS	470 1,696	361 1,300	474 1,712	358 1 294	474 1,712	353 1,294	716 2,588	859 3,087	716 2,588	859 3,087	470 1,696	382 1,371	6,497 23,425
II-9 Near IR Spectrometer Mini Mount Pump	572 1,606 1 376	503 1,576 1,376	572 1,610 1,376	503 1,576 1,376	572 1,610 1,376	503 1,576 1,376	957 1 762 1,376	1,081 1,818 1,376	957 1,762 1 376	1,081 1,818 1 376	709 1,650 1,376	463 860 1,376	8,473 19,224 16,512
II-10 Cryo Cooled Interfer Spec.	336	264	336	252	336	252	504	637	504	637	336	217	4,611
I-21 Gas Release System	ALL POWER SELF CONTAINED EXCEPT THERMAL WHICH IS COVERED BELOW												-
II-3 OBIPS on Pointing Platform Mini Mount	72 123	- -	72 123	- -	72 123	- -	72 123	- -	72 123	- -	72 123	- -	432 738
I-9 Electron Accelerator	1,719	6,699	1.719	4,980	1,719	4,980	-	-	-	-	-	-	21,816
III-3 Level I Beam Diagnostics	10	6	10	6	10	6	-	-	-	-	-	-	48
III-2 Vector Magnetometer III-4 Level II Beam Diagnostics II-3 OBIPS on RMS	SAME AS I-21												
III-25, III-18, III-23 ESP	same as I-21												
IV-1 Solar Flux Cal Array	-	7	-	-	-	-	-	7	-	7	-	6	27
ICCM	4,800	4,800	4 800	4,800	4,800	4,800	4,800	4,800	4,800	4,800	4,000	4,800	57,600
Thermal	3 307	3,197	3,315	3,108	3,315	3,108	3,456	3,456	3,456	3,456	3,456	3,186	39,816
Lidar Pump	1,008	1,008	1,008	1,008	1,008	1,008	1,008	1,008	1,008	1,008	1,008	1,008	12,096
Freon Pump	3,288	3,288	3,288	3,288	3,288	3,288	3,288	3,288	3,288	3,288	3,288	3,288	39,456
	29,428	33,405	29,460	29,374	29,460	29,374	34,300	37,262	34,300	37,262	30,784	23,712	377,621

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instrument modules that are deployed from the Spacelab pallets. Three options were identified: 1) use of an RF link with receiving, transmitting, and power equipment included in the instrument package design; 2) hardwiring from the pallet to the instrument package by routing along and attaching to the Shuttle Remote Manipulator System (RMS); and 3) hardwiring from the pallet to the instrument package using a cable management system mounted on the pallet to extend and retract cabling as instruments are deployed and recovered.

Hardwiring via the RMS was not selected due to the increased inter-center interface effort, the increase in line lengths, and the need to design an RMS End Effector with electrical interfacing capabilities. Hardwiring using a cable management system was reviewed and eliminated due to the many technical problems associated with the design, build and installation of the cable management system. The RF link was selected as the method most readily adaptable to the operation of deployed instruments. Although there would be more deployed weight, the technical problems associated with interfacing, design and installation of the link would be minimal when compared to either of the two hardwired methods. All RF and EPDS components required are available off-the-shelf, resulting in the most cost effective solution to the problem. Details of the trade study are included in Section 5.4.1.

<u>Deployed Instrument Power Supply Analysis</u> - Each of the deployed instrument packages, both free-flying and RMS maneuvered, requires an internal power supply to meet the energy requirements of the instrument and the supporting subsystems. This analysis evaluated the power and energy requirements for each such package in the Flight 1 and Flight 2 configurations and selected the battery required to satisfy the requirement. A power and energy margin of 100% was used as a goal to insure adequate margin for growth and contingency operation. Such a margin is considered necessary during the preliminary design phase to minimize iteration to the EPDS design resulting from changing power requirements and operational procedures. This analysis will be updated during the hardware design phase using smaller power margins which may result in reduced component sizes. The details of the analysis are available in Section 5.4.2.

4.4 Attitude and Pointing Control Subsystem (APCS)

The following sections describe the instrument pointing requirements as initially defined in the IFRDs, amended as appropriate, and summarize the APCS configuration selected for the AMPS flights. The Orbiter's attitude control capability was investigated, and where found lacking, appropriate recommendations were made to augment the basic capability of the Orbiter. The additional equipment required consists of two types of pointing platforms with appropriate sensors, all of which are assumed to be furnished by the government (GFE). Supporting analyses and descriptions are included herein to ascertain the capability of these platforms to meet the pointing requirements.

4.4.1 Attitude and Pointing Control Requirements

Flight 1 - The instruments have been grouped as to their particular mission task as delineated in Table 4.4-1. The pointing accuracy and stability (tabulated as stability rate) requirements were obtained initially from the Instrument Functional Requirements Document (IFRD) and amended as necessary by cognizant personnel for clarification and consistency. Values stated were continually under investigation and represent an interpretation of the requirements based on the latest available data.

The nominal vehicle attitude is X-POP/Z-LV. II-9 (NIR Spectrometer) requires pointing toward the sun during the sunrise and sunset terminators for approximately 3 minute periods. II-7 (Cryo Limb Scanner) and II-10 (Cryo Cooled Interferometer/Spectrometer) have been located on the SIPS due to the $\leq$ 0.5 degree co-alignment requirement between instruments and require horizon pointing. It is assumed that II-7 will limb scan with an internal mirror. II-7 requires that data obtained from individual samples be accurate to each other sample within the scan angle $\pm$ 4 arc seconds. (See Table 4.4-1.)

II-3 (OBIPS) concerns itself only with the gas dynamics portion of the Acoustical Gravity Waves experiment. With the vehicle in the X-POP/Z-LV attitude, a roll maneuver about the X-axis is initiated. At the proper position (Z-axis along the velocity vector), the chemical release canister is ejected. The vehicle then rolls to the optimum viewing orientation. At approximately 80 km back in the orbital plane, OBIPS will view the anticipated 10 km diameter gas cloud. OBIPS will view the cloud just prior to release until 5 seconds after release. At the 80 km distance, the anticipated cloud diameter will be within approximately 7 degrees of the total 16 degree FOV of OBIPS. Therefore, $\pm$ 1 degree pointing accuracy was deemed sufficient.

The remainder of the instruments are either hardmounted, free-flying or deployed by the Remote Manipulator System (RMS) and no problems are anticipated in meeting their pointing requirements.

Table 4.4-1 Flight 1 Pointing Requirements

Mission Task	IFRD No.	PTG Acc, ± deg	Stability Rate, deg/sec	Remarks
Minor Const. Profiles	I-1	1 (ZLV)	0.005 (⊥OA; 0.01 (↻OA)	Harmounted LIDAR
	II-9	0.25 (Sun occult)	0.1/2 min (B-Axes)	MPM No. 1 with thermal canister; internal solar tracker.
	II-7	0.5 (Horizon)	0.003 (⊥Hor); 0.1 (↻OA)	SIPS yoke A; int scan mirror;rel sample loc ± 4 sec (knowledge) (≤0.5 deg Co-align)
	II-10	0.25 (Horizon)	0.001/10 sec	SIPS yoke B (≤0.5 deg Co-align)
Acoust. Grav. Waves	I-21	3 (VV)	N/A	Hardmounted chemical release (No. 1).
	II-3	1	0.1	OBIPS/sunshade on MPM No. 2; gas dynamics portion of experiment
Beam Inter 1	I-9	10 ($\overline{B}$-Field)	N/A	Hardmounted elect. acc.; vehicle pointing under III-2 control; 2 deg att knowl of $\overline{B}$-Field
	III-3	10 ($\overline{B}$-Field)		Hardmounted gas release (level I beam diag) to I-9
	II-3	2	1	OBIPS W/O Sunshade (RMS-Located)
	III-2	2 ABS	N/A	Vector magnetometer (± 0.5 deg instr acc) (RMS-Located)
Beam Inter 2	III-4	See Remarks	N/A	Level II Beam Diag; pos acc 0.1M (B-Axes) (RMS-Located)
EMI Field Map	III-25	See Remarks	N/A	EMI Diagnostic Packages; spin axis knowl to ± 5 deg of vehicle X-axis; pos acc ± 0.1m (B-axes) (RMS-Located)
Contami-ation	--	N/A	N/A	Integ env contam mon hardmounted
Solar Monitor	IV-1	1 (See Remarks)	Free Drift	Abs solar flux cal. pkg hardmounted. Internal gimbal system; Orbiter in free drift mode (ptg. acc. at data take initiation.)
Wake Mapping	III-25	1 (See Remarks)	N/A	EMI Diagnostic Packages (RMS-Located)
	III-18	1 (See Remarks)		Planar RPA (RMS-Located)
	III-23	1 (See Remarks)		Neutral Mass Spect (Ptg acc same as EMI Map) (RMS-Located)

Flight 2 - The instruments have been grouped as to their particular mission task as delineated in Table 4.4-2. The instruments which require supplemental pointing capability are nearly identical with those described for Flight 1. II-4 (UV-VIS-NIR Spectrometer/Photometer) has been added to the forward pointing platform. This instrument requires horizon pointing and the platform can be time-shared with the II-9 instrument which requires sun pointing at terminators only.

With the exception of II-3, the remainder of the instruments are either hardmounted, deployed by the RMS, or free-flying.

Table 4.4-2 Flight 2 Pointing Requirements

Mission Task	IFRD No.	PTG Acc, ± deg	Stability Rate, deg/sec	Remarks
Minor Constituent Profiles	I-1 II-7 II-10 II-9	Same as Flight 1		
	II-4	0.1 (ABS)	0.003/1 sec	UV-VIS-NIR Spect/Photometer } MPM No. 1 with Modified SIPS Thermal Canister
Conductivity Mod	I-21 II-3	3 (VV) 1 (Orb Object)	0.017 0.1	Hardmounted Chemical Release Sys (No. 2) OBIPS/Sunshade on MPM No. 2; view release, release zone on next pass
Wave/Particle Inter. A	I-12 III-2	5 ($\vec{B}$-Field) 2 (ABS)	0.01/10 min N/A	Plasma Wave Package (XMTR, RCV, Ant) Hardmounted Vector Magnetometer on RMS No. 1; Accuracy Knowledge 0.01 deg.
Wave/Particle Inter. B Long Delay Echo	I-12 III-2	Same as Wave/Part. Interaction A		Receiver & Antenna on Free-Flying RF Rcv. Pkg. (Includes Magnetometer for Att Ref)
Plasma Flow	III-17	N/A	0.25	Deployable Test Body on RMS No. 2; Position Knowledge 0.1m - 3 Axes
	III-2 III-10 III-18 III-22 III-23	10 (VV)	N/A	Vector Magnetometer Ion Mass & Dist Anan Planar RPA; Allowable Rates ≤2°/sec Langmuir Type Current Collector Neutral Mass Spect. Position Acc ± 0.1m (3-Axes) } RMS No. 1
Solar Flux Monitor	IV-1	5	Free Drift	Same as Flight 1

4.4.2 Orbiter/Spacelab Pointing Accuracy Capabilities

Table 4.4-3 summarizes the current Orbiter/Spacelab pointing capabilities for the various reference modes as defined in Volume XIV, Rev. D "Space Shuttle Payload Accommodations" document. Using attitude deadbands of ± 0.1 deg/axis, the pointing accuracy at the IMU navigation base located forward of the crew compartment is on the order of ± 0.5 degrees. Errors that contribute to this 3 σ value include, in addition to the attitude deadband, IMU drift, gimbal readout inaccuracy, and flight control system errors. At the payload line-of-sight (LOS), however, the pointing accuracy is further degraded due to structural misalignment, vehicle flexibility and thermal deformations to a value in excess of 2 degrees. This is referred to as open loop control of the payload LOS. In order to achieve comparable pointing accuracies at the payload LOS as at the Orbiter navigation base, attitude data from a payload-mounted sensor is fed back

to the Orbiter GN&C computer. The computer has existing interface capability to accept this data. In this manner, pointing accuracies at the payload LOS are commensurate with those at the IMU navigation base as indicated in the table.

Table 4.4-3 Orbiter/Spacelab Pointing Accuracy Capabilities

Reference Mode (1)	IMU Only (Open Loop)		Payload Sensor Feedback
	At IMU, deg - 3σ	At Payload LOS, deg	At Payload LOS, deg - 3σ
Inertial	$\pm$ 0.5	2	$\pm$ 0.5
Celestial	$\pm$ 0.44	2	$\pm$ 0.44
Earth Fixed Target (2)	$\pm$ 0.5	2	$\pm$ 0.5
Local Vertical (2)	$\pm$ 0.5	2	$\pm$ 0.5

Note: (1) Orbiter Vernier RCS Deadband: $\pm$ 0.1 deg/Axis

(2) Based on 100 nm Orbit; TDRS Navigation Uncertainty = $\pm$0.28 deg

4.4.3 Attitude and Pointing Control Concepts

Pointing platforms have been defined to augment the Orbiter pointing capabilities principally in three ways.

- o They are used for precision pointing of instruments which require greater accuracy and stability than the Orbiter can provide.
- o They extend the viewing range of instruments with little or no Orbiter maneuvers required, thereby minimizing propellant usage and reducing potential contamination concerns.
- o They allow observation of various targets simultaneously.

The configuration for Flight 1 is shown in Figure 4.4-1. An evaluation of pointing platforms, as discussed in Section 5.2.1, led to the selection of two types for use on AMPS: Miniaturized Pointing Mount (MPM) and Small Instrument Pointing Systems (SIPS).

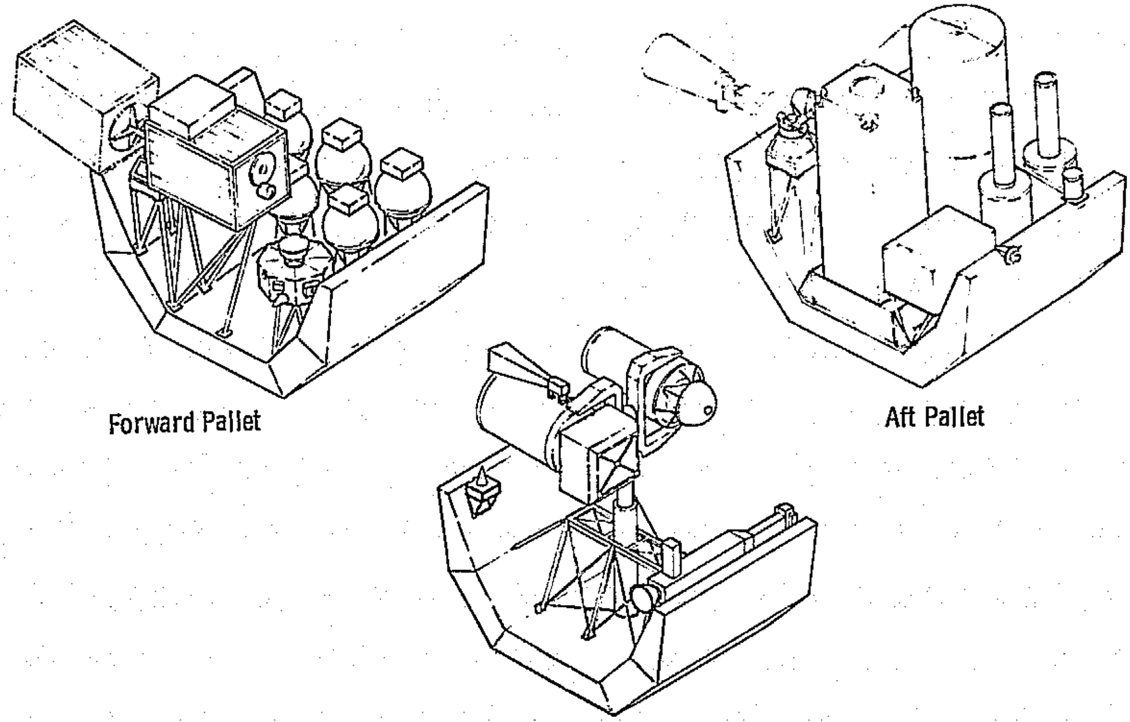


Figure 4.4-1 Pointing Platform Configurations

The forward pallet contains a 3-gyro MPM with an "ATM-type" thermal canister containing II-9 (Near IR Spectrometer). The canister is a preliminary design of an environmental enclosure utilizing spare thermal control equipment from ATM. A NASA Standard Fixed-Head Star Tracker (FHST) is located on the MPM.

The center pallet contains a SIPS, which is the baseline pointing system, without thermal canisters. One yoke contains II-7 (Cryo-Cooled Limb Scanner) while the second yoke contains II-10 (Cryo-Cooled IR Spectrometer). Separate gimbal rings which surround each instrument are attached to the yokes and contain a two-gyro package each. A second NASA Standard FHST is located on II-7.

The aft pallet contains a second three-gyro MPM with II-3 (OBIPS with sunshade).

The Flight 2 pointing platform configuration is essentially identical as for Flight 1. The differences are that a modified SIPS thermal canister has been substituted for the ATM-type and an additional instrument, II-4 (UV-VIS-NIR Spectrometer/Photometer), has been included on the forward pallet MPM.

SIPS Description. - The block diagram of Figure 4.4-2 indicates a two-axis gyro-stabilized control system for each yoke of the SIPS. Separate rate-integrating gyros for the elevation (up/down) and the left/right axes provide stabilization signals in the two control loops, respectively. When a gyro senses a disturbance about its input axis, it begins to precess through gyroscopic action. As it precesses, its signal generator pickoff sends an error signal to the Control Electronics where the signal is compensated, amplified, and routed to the respective gimbal DC torquer. The torquer proceeds to drive the gimbal

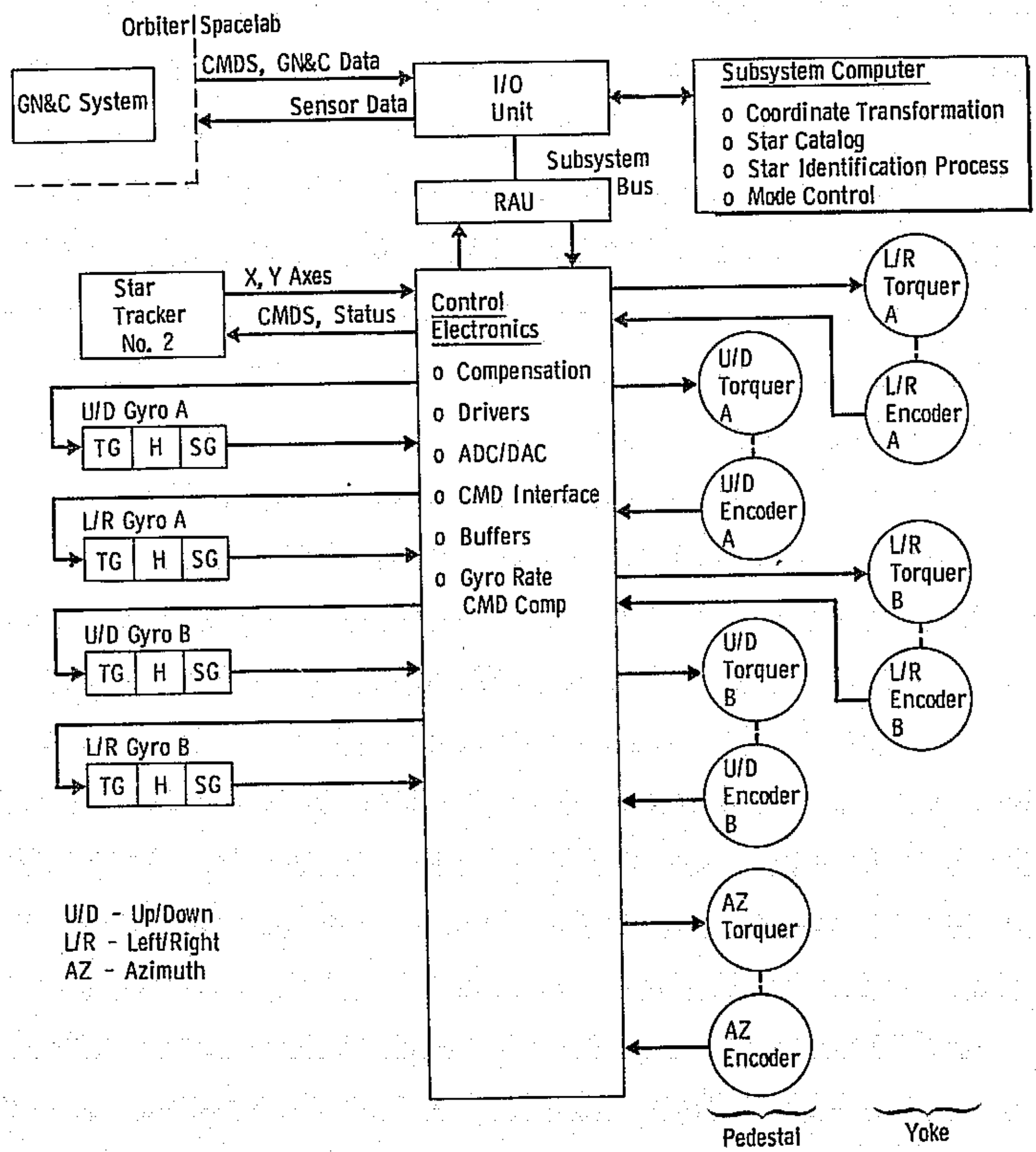


Figure 4.4-2 SIPS Functional Block Diagram

in the proper direction so as to null out the disturbance. The gyro pickoff output signal, in the form of a discrete pulse wavetrain, also serves as a measure of the attitude change during the time period of interest. Position commands from the crew drive the azimuth loop (pedestal) in an open loop fashion. The FHST provides attitude data to the Orbiter GN&C computer, is used for platform alignment, and for updating gyro drift.

The telescoping pedestal indicated in Figure 4.4-3 allows both yokes to be deployed simultaneously from the stowed position; i.e., along the longitudinal axis of the Orbiter. A ball screw is driven by a pair of redundant brush-type torquers through a gear train to effect deployment. A brake is used to hold the deployed pedestal. In an emergency, a retraction reel/tension spring system retracts the pedestal. If both systems fail and the payload has been deployed past the Orbiter mold line, payload separation is implemented by a jettison device at the upper portion of the pedestal. A brush-type DC torquer rotates both yokes simultanesouly in azimuth; a twelve-bit encoder allows gimbal position readout. The elevation or up/down axis and the left/right axis provide fine pointing of the instruments located within the two yokes. Each fine control axis contains a limited rotation brushless DC torquer (10.6 N-m peak torque) and a twelve-bit encoder. Gimbal freedom provisions are as follows:

Azimuth (Z- Axis)	-	$\pm$ 200 degrees
Elevation (Up/Down)	-	0 to 120 degrees
Right/Left	-	$\pm$ 10 degrees (minimum)

The current weight estimate for the SIPS including two canister gimbal frames and the deletion of the two canister assemblies is 618 kg. Each yoke can presently support 500 kg (instrument plus canister) and accommodates a canister size of 1 x 1 x 3 meters.

<u>MPM Description</u> - The MPM implementation, shown in Figure 4.4-4, utilizes three rate gyros in a strapdown configuration as part of the attitude determination system. The three gyro outputs are sent to the Control Electronics where the strapdown computation is performed by means of quaternions which relate the desired reference frame with respect to the instrument reference frame. The instrument attitude and attitude rate are combined in the control law which is assumed to consist of instrument rate, position and the integral of position. The output of the control law is sent to the gimbal steering law whose output, in turn, sends gimbal torque commands to the appropriate torquer. The steering law output is also a function of the gimbal positions as indicated by the resolver/encoder outputs. For the forward MPM, the FHST provides the same functions as described for the SIPS and is used to supplement the solar tracker (internal to II-9-NIR Spectrometer) during periods of sun occultation when an update is required. For Flight 2, an additional instrument (II-4, UV/VIS/NIR Spectrometer) which

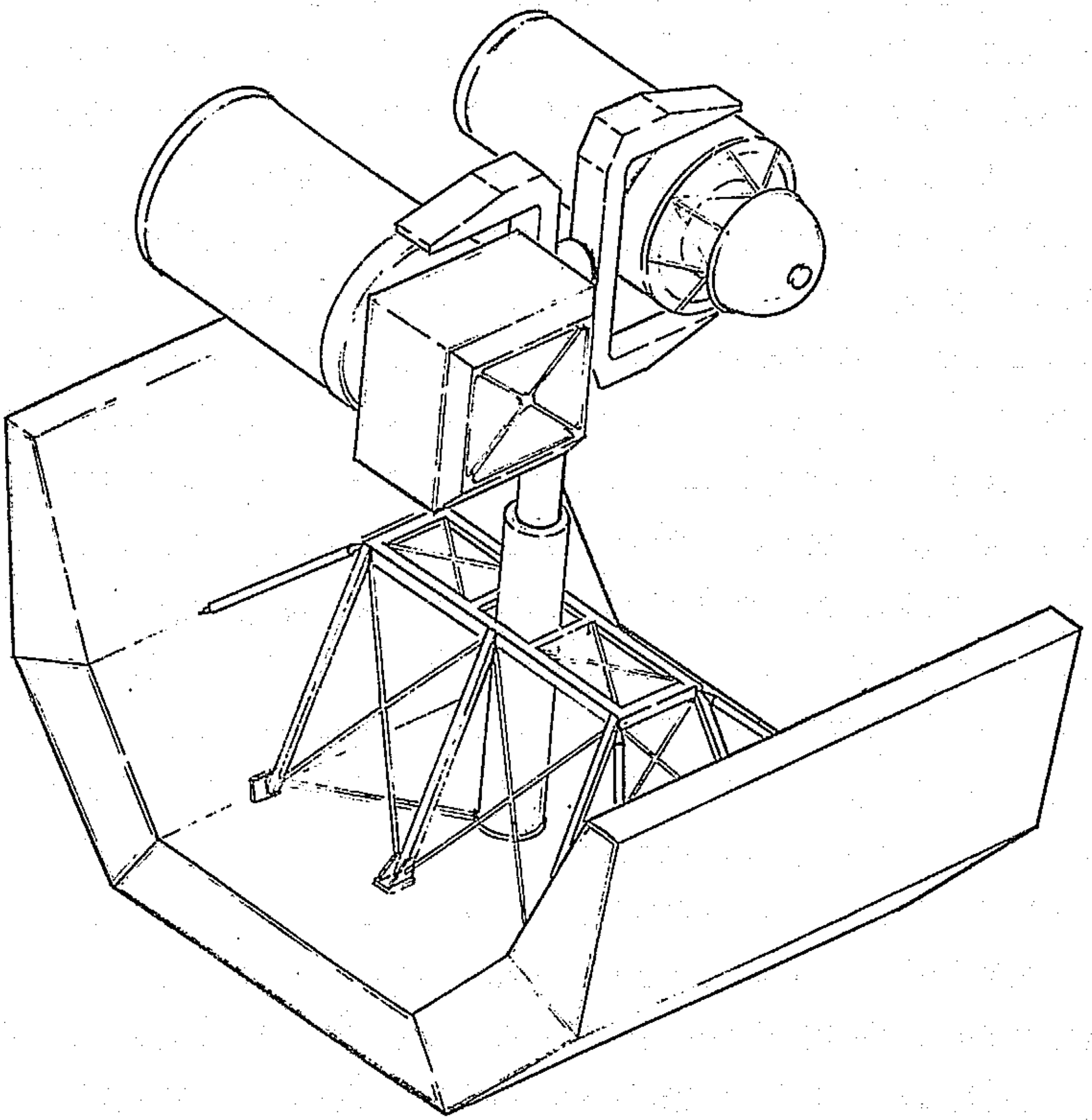


Figure 4.4-3 Small Instrument Pointing System

requires earth-viewing is located within the thermal canister. This flight will require more frequent attitude reference system updates from the FHST or sun sensor as the instruments are alternately pointed at the sun or the earth.

The MPM is an Inside-Out Gimbal system with suspension between the pedestal and pallet floor. Figure 4.4-5 depicts a conceptual design of the MPM with overall dimensions and gimbal angular freedom as noted.

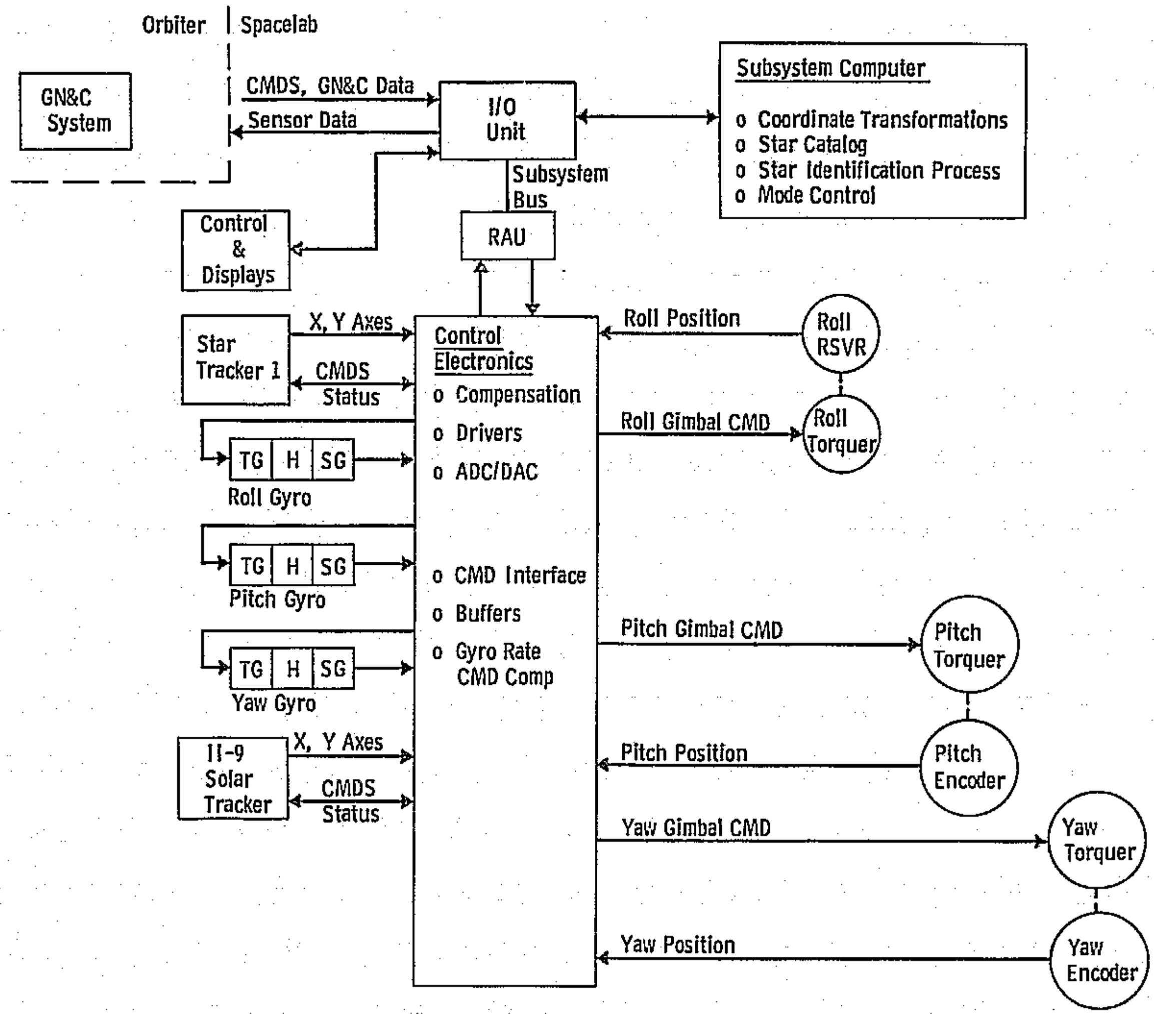


Figure 4.4-4 MFM Functional Block Diagram

Currently, the pitch and yaw axes each contains a brush-type DC torquer (0.64 N-m peak torque) and a 15-bit encoder. Each of these axes also contains a DC tachometer, from which gimbal rate information may be derived. The add-on roll axis capability will contain the DC torquer as the other two axes in addition to a single-speed resolver. The resolver will be used for gimbal position readout. The outputs of the resolver and the two encoders are used in the torquer control law computations.

The mount isolation system has three translational and three rotational degrees-of-freedom. The suspension characteristics (stiffness and damping) plus adequate control loop bandwidth accommodate the large

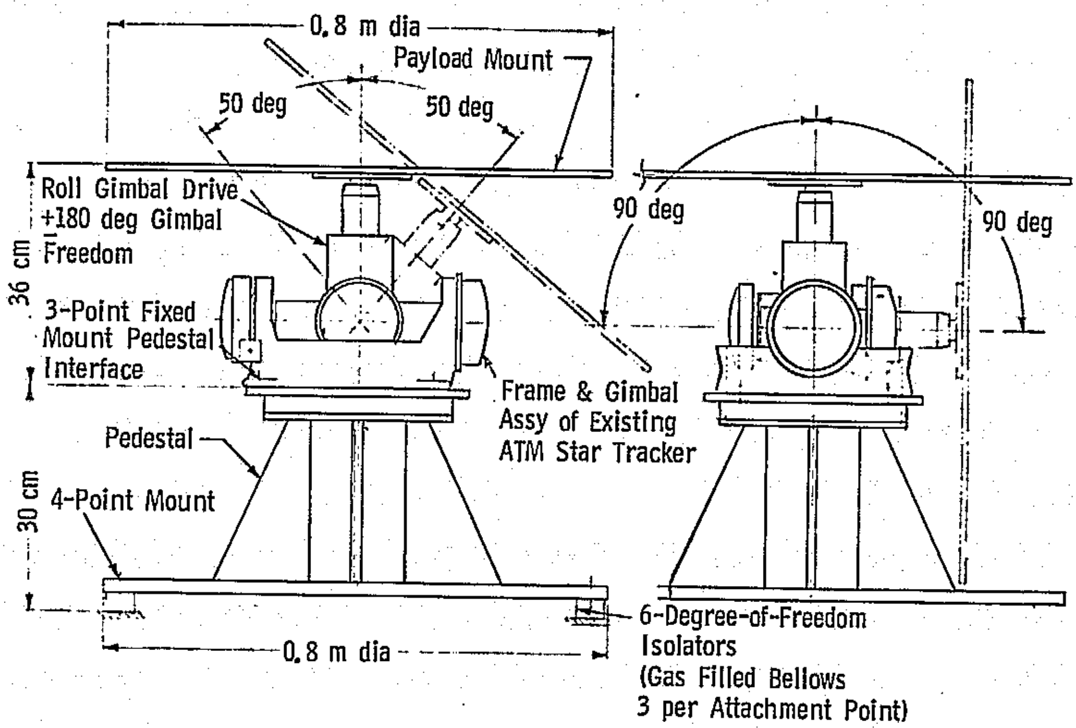


Figure 4.4-5 Mini-Mount Platform

center-of-mass offset, i.e., no mass balance of the instrument is required. However, the payload and gimbal mount each must be caged separately during the launch and landing phases of the mission. This is accomplished by having the instrument baseplate temporarily disengage from the mount and separately clamp the mount and instrument canister to the pallet floor.

Present weight estimate of the MPM without a thermal canister is 56 kg. The platform can accommodate a 1 (diameter) x 3 meter canister and a total payload weight of 500 to 600 kg.

4.4.4 Rationale for Configuration

As previously described, pointing platforms are used to supplement the basic Orbiter capability. The assumption was made that both the SIPS and MPM will be GFP and that they will be available at the proper time frame. In order to determine if AMPS requirements could be met, it was imperative that these fine pointing and stabilization systems be examined in depth to ascertain their capabilities. Accordingly, detailed digital simulation studies of both pointing platforms were conducted for pointing and tracking modes. The latter mode tracked a fixed point on earth directly below the orbiting vehicle for an or-

bital period 5 seconds prior to and 5 seconds after the nadir point; the former examined perturbations about an inertially-fixed point. The tracking sequence was chosen in order to impose maximum rates and accelerations on the platforms during an earth tracking operation. The model used and results from the analysis of these two modes are discussed in Sections 5.5 through 5.5.4. The three inertial-grade rate gyros utilized in the study, 64 PMRIG, LDG 540 and Gyroflex, are some of the candidate sensors for the NASA Standard Inertial Reference Unit. Moreover, the noise Power Spectral Density characteristics of these gyros were readily available and were used for the sake of expediency. The gimbal pivot friction model used for the SIPS study represented the major error source characteristic of limited rotation brushless type DC torquers, i.e., motor magnetic hysteresis. A standard Dahl friction model (Reference: AIAA Paper No. 75-1104, August 1975) was used to simulate this phenomena. For the MPM study a Dahl friction model was simulated which included the effects of bearing and brush friction, and motor magnetic hysteresis. The external disturbance to the pointing platform used in the pointing mode studies was crew wall-pushoff. This type disturbance has been used in other current pointing studies (Experiment Pointing Mount Working Group under JPL auspices) and is anticipated to be the most severe crew motion type disturbance. It is also noted that the impulse (44.5 N-s) imparted by the vernier RCS thruster firings to the pointing platforms had essentially the same effect as crew motion disturbance upon pointing stability. The nominal system, for the pointing studies, consists of the control loop bandwidth used (1 or 5 Hz); a "heavy" instrument mass (500 kg); a soft isolation system ($\zeta = 0.01$)--for the MPM; and the instrument points straight up (+Z-direction).

A brief summary of the salient conclusions derived from the pointing and tracking studies is listed below.

(1) Pointing Mode Performance

General

- Both SIPS and MPM will satisfy AMPS requirements;
- For control loop bandwidths of 1 or 5 Hz, maximum stability error recorded (SIPS and MPM) is in the subarc-second range for crew motion, rate gyro noise, and friction levels investigated;
- Higher bandwidth (5 Hz) reduces stability error for either platform.

SIPS

- o With no friction, stability error is essentially constant for various gyro noise levels;
- o With friction included, stability error is higher but essentially constant for various gyro noise levels;
- o Gyro noise level has minor effect on pointing stability.

MPM

- o With no friction, stability error is essentially constant for various gyro noise levels;
- o With friction included, stability error increases slightly but is essentially constant for various gyro noise levels;
- o Friction level has minor effect on pointing stability.

(2) Tracking Mode Performance

General

- o Both SIPS and MPM will satisfy AMPS requirements;
- o For control loop bandwidth investigated (1 Hz), maximum stability error recorded is in arc-second range for no crew motion, rate gyro noise, and friction levels investigated.

SIPS

- o With no friction, stability error is essentially constant for various gyro noise levels;
- o With friction included, stability error is higher but essentially constant for various gyro noise levels.

MPM

- o With or without friction, stability error is essentially constant for various gyro noise levels;
- o Friction level studied has no effect on pointing stability.

In addition to the aforementioned pointing studies, investigations were conducted to determine a preliminary static error budget and the implementation of an attitude reference system. A static error model was derived which considered the major error sources of the components and their effects on the static pointing accuracy. The model and results are contained in Section 5.5.5. The attitude reference system chosen utilizes a moving reference frame approach that allows both maneuvering, such as earth pointing, and a smooth transition from one inertial pointing orientation to another. The implementation is described in Section 5.5.6.

4.4.5 Sensor Considerations

FHST for Limb Viewing Instruments - The question has been raised as to the feasibility of obtaining update orientation data with a NASA Standard Fixed Head Star Tracker (FHST) mounted in the SIPS together with the AMPS instruments designed for limb viewing. This could become an integration problem if stars could not be tracked when the Orbiter is in the Z-LV mode. Several factors must be considered:

(1) Celestial viewing limitations generated by the Orbiter structures in the upward direction, and the earth and lower atmosphere in the downward direction;

(2) Performance-deteriorating effects on the FHST due to the sunlit earth or Orbiter structures, or direct sunshine when viewing toward the sunset direction;

(3) Timelines for continuously-changing guide-star fields.

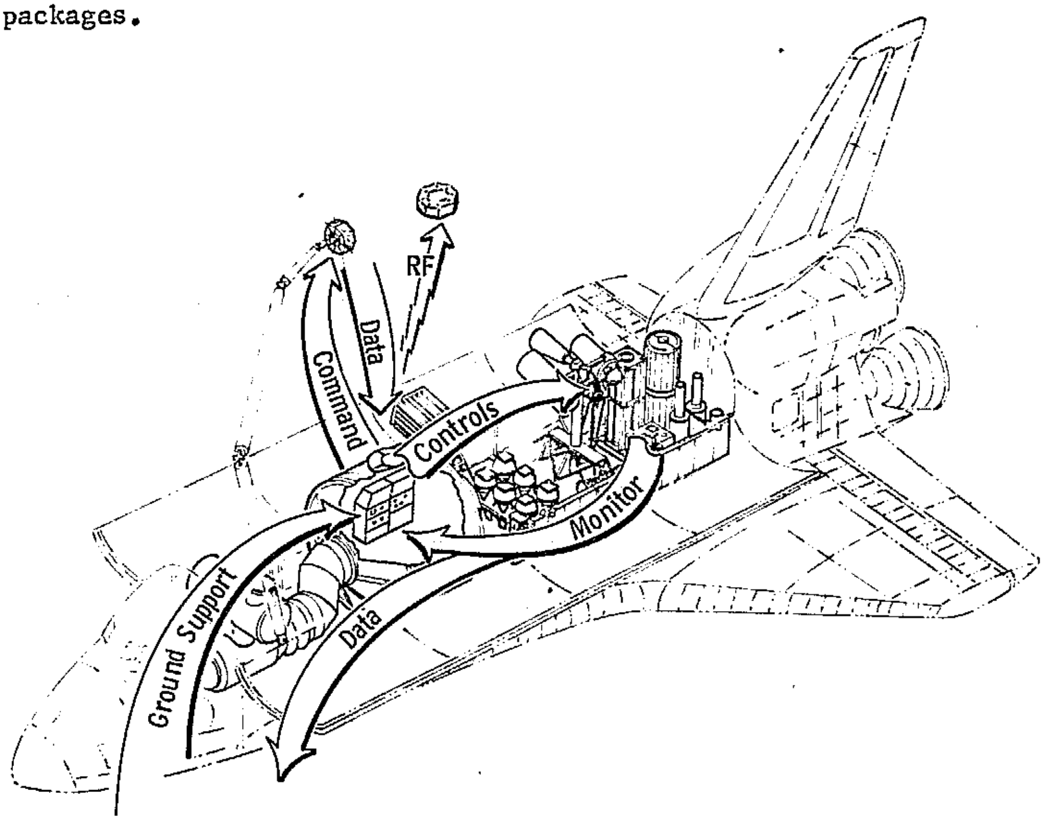
Preliminary analysis indicates that, for worst case star field conditions, the star tracker can provide the necessary update information. Section 5.5.7 discusses the results and conclusions. The star tracker line of sight must be skewed with respect to the instrument line of sight in order to optimize the look angle between the earth and Orbiter structure.

FHST for Cloud Viewing Instruments - An FHST has not been included on the MPM located on the rear pallet. This MPM contains the OBIPS instrument whose pointing accuracy requirements are in the coarse range ($\pm$ 1 degree). It is felt that the FHST located on the SIPS (middle or adjacent pallet) can be used in the attitude determination system of the rear MPM. Should, however. the structural misalignments and thermal deformations prove to be excessive (e.g., from Orbital Flight Tests) an additional FHST would be required. Alternately, for a more conservative design, an FHST would be added if available from inventory.

4.5 Data Management Subsystem (DMS)

4.5.1 Data Management Requirements

The science requirements imposed on the AMPS DMS (as defined in the IFRDs) reveal a broad spectrum of data, diverse instrument control and monitor functions, further impacted by the need to interface with deployed packages.



Compilation of Flights 1 and 2 data and command requirements (see Tables 4.5-1 and 4.5-2) reveal digital data may be as high as 7.6 Mbps (Flight 2), while analog data bandwidths in excess of 1 MHz and video data of 4.2 MHz are required during the mission. Control functions to an instrument will include discretes and serial commands which for Flight 1 will require at least 376 discretes and 54 serial commands. However, the data and command requirements at any one time are a function of the experiment mission timelines.

Figure 4.5-1 provides a typical data profile for day one of Flight 1. Digital data requirements are either in the 100 Kbps range or in excess of 6 Mbps. Simultaneous recovery of digital, video and analog data is also required during the electron beam experiment. An overall data profile for a six day flight is shown in Figure 4.5-2. The extensive operation of the minor constituent experiment is reflected in the high data rates of 2.5 to 6.3 Mbps. While video data acquisition

Table 4.5-1 Flight 1 Data and Command Requirements

Instr.	Digital RAU Input	Digital Hi-Rate MUX Input	Analog	Video	Commands Discrete	Commands Serial
Minor Constituent						
I-1 Laser Sounder	11 kbps				50	12
II-7 Cryo Cooled Limb Scanner	0.5 kbps	11 kbps			6	5
Platforms	3.5 kbps					
II-9 Near IR Spectrometer	0.6 kbps	3.8 mbps			6	4
II-10 Cryo-Cooled Interf spec.	0.5 kbps	2.5 mbps			6	4
Acoustic Gravity						
I-21 Gas Release (Deployed)	0.1 kbps				84	6
II-3 OBIPS	12 kbps			4.2 mhz	36	6
Ptg. Platform	1 kbps					
Electron Beam Study						
I-9 Electron Accelerator	5 kbps		pulses(6) 1 MHz (2)		30	4
III-3 Level 1 Beam Diagnostic (Gas plume)	4 kbps				4	2
II-3 OBIPS	12 kbps			4.2 MHz	36	6
Beam Diagnostic Pkg (on RMS)						
III-2 Vector Magnetometer	3 kbps				16	-
III-4 Level II Beam Diag	82 kbps		1 KHz (2) 100 KHz (5)		20	4
Env. Sensing Pkg.						
(on RMS & deployed)	16 kbps				88	6
Solar Flux	13 kbps					

Table 4.5-2 Flight 2 Data and Command Requirements

Instr.	Digital: RAU Input	Digital: Hi-Rate MUX Input	Analog	Video	Command: Discrete	Command: Serial
Minor Constituent						
I-1 Laser Sounder	11 kbps				50	12
II-4 UV-VIS-NIR Spect/ Photometer	10 kbps	1 mbps			64	7
II-7 Cryo-Cooled Limb Scanner	0.5 kbps	11 kbps			6	5
II-9 Near IR Spectrometer	0.6 kbps	3.8 mbps			6	4
II-10 Cryo-Cooled Interf Spectrometer	0.5 kbps	2.5 mbps			6	4
Chemical Release						
I-21 Chemical Release (Deployed)	-				14	
II-3 OBIPS	12 kbps			4.2 MHz	36	6
Platform	1 kbps					
Wave Particle/Delay Echo						
I-12 Transmitter/Recvr	0.3 kbps		30 khz		30	3
III-2 Fluxgate Vector Magnetometer	3 kbps				16	
RF Receiver Package (on RMS)						
I-12 RF RCVR	0.3 kbps		30 KHz		4	3
III-2 Fluxgate Vector Mag.	3 kbps				16	0
Plasma Flow						
Deployable Test Body	1 kbps				12	
Diagnostic Package	90 kbps	-	1 KHz (3)		108	7
III-2 Fluxgate Vector Mag.						
III-10 Ion Mass Distri. Anal.						
III-18 Planar RPA						
III-22 Langmuir Probe						
III-23 Neut. Mass Spect.						
Solar Flux	13 kbps					

requirements are not extensive, the need for simultaneous acquisition of video, analog and low rate digital data may require substantial RF bandwidth. The figures indicate that, in general, the data acquisition can be viewed in terms of separate experiments operating in sequence. This data together with the communication coverage shown in Section 5.8.1 will determine when data can be transmitted in real time or must be recorded.

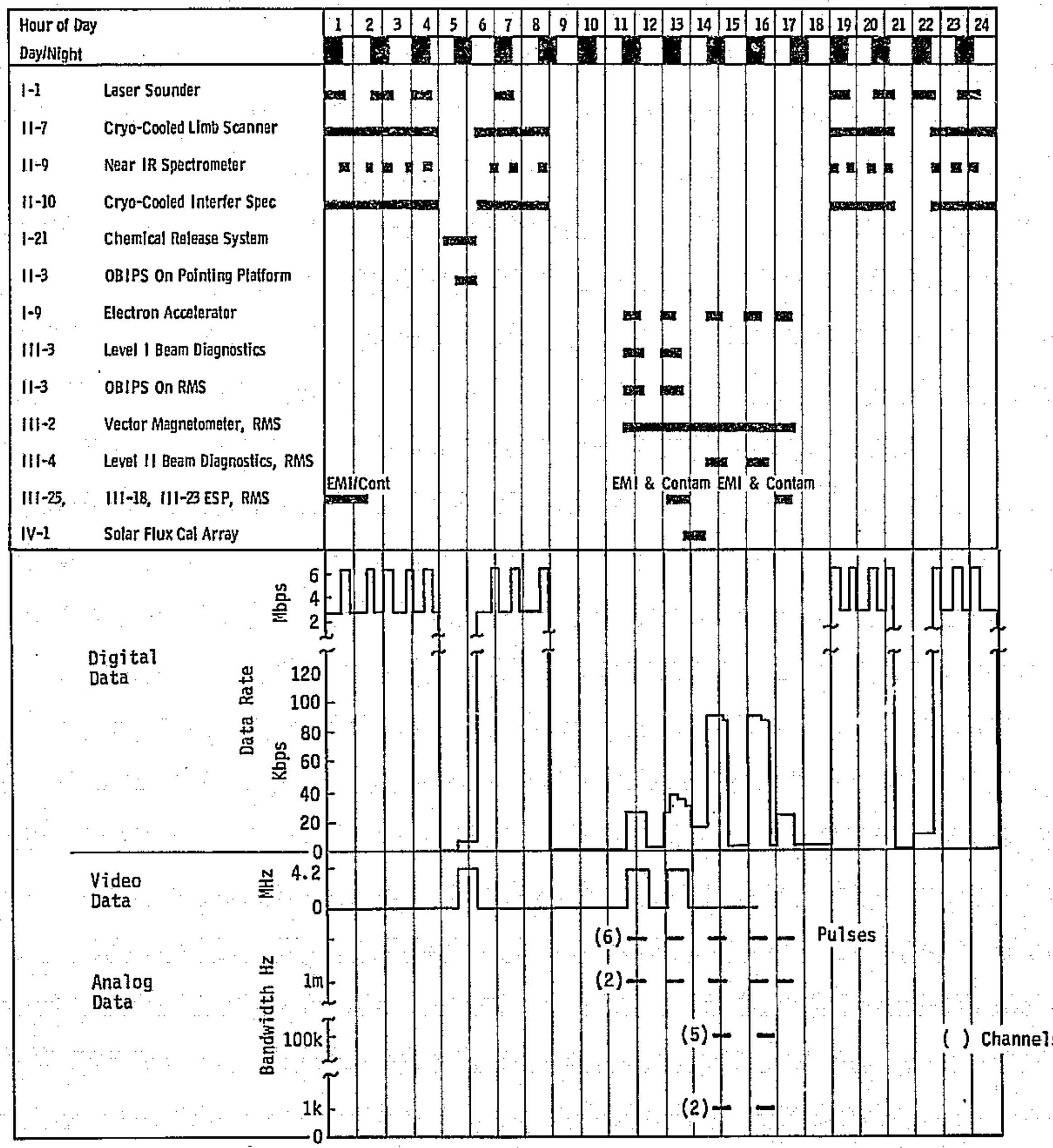


Figure 4.5-1 Instrument Timeline, Day 1 - Flight 1 Data Profile

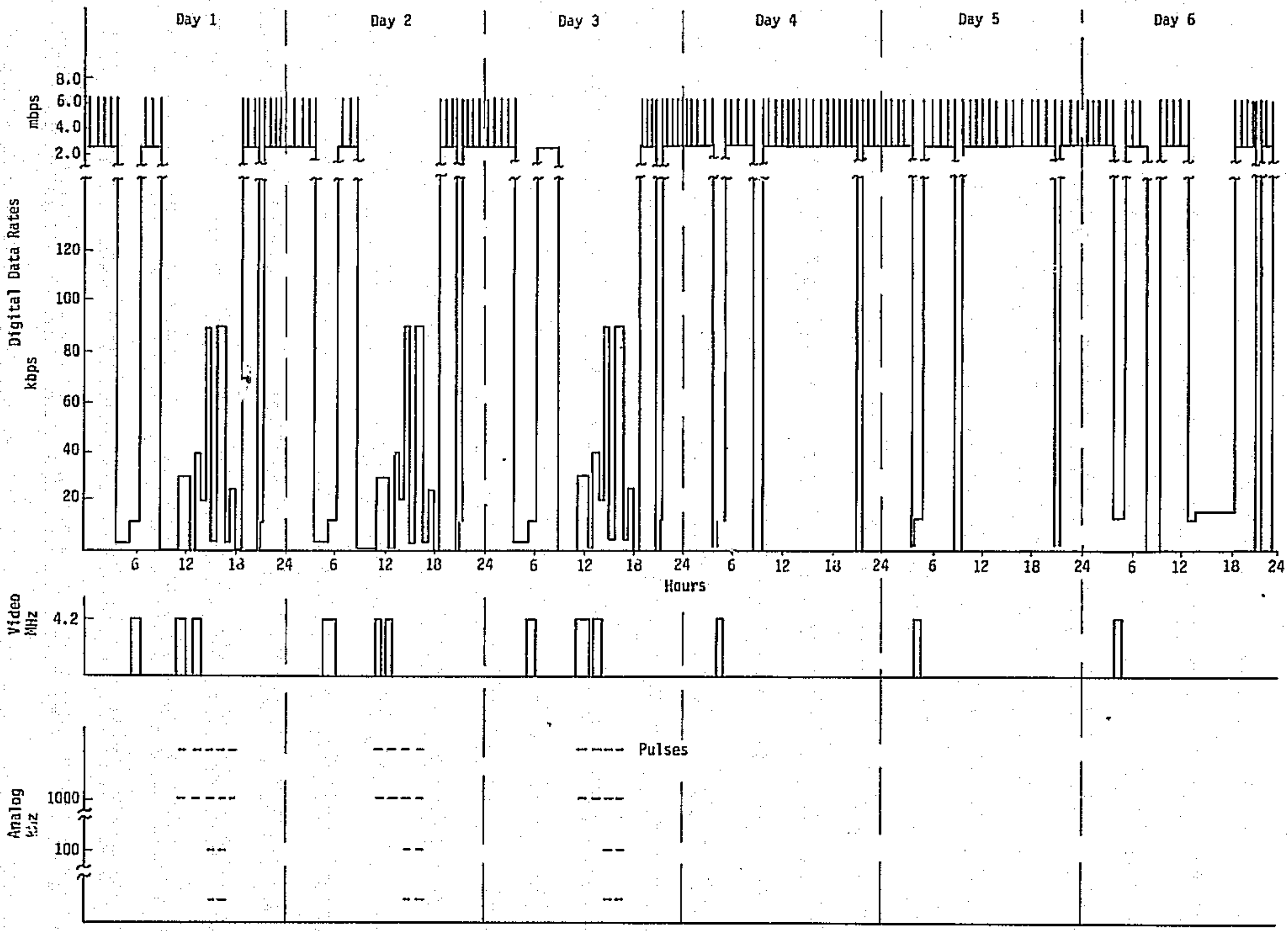


Figure 4.5-2 Flight 1 Mission Six Day Data Profile

4.5.2 Data Management Concepts

The most important factor in the system design of the AMPS DMS is to maximize the use of existing Orbiter/Spacelab capabilities and minimize new hardware development. Our design approach included a review and analysis of the Spacelab capabilities as identified in the Spacelab Payload Accommodation Handbook and summarized in Figure 4.5-3. The resulting baseline AMPS DMS is shown in Figure 4.5-4. In addition, the AMPS payload requires dedicated DMS hardware for diagnostic packages which are mounted on the ends of booms or are deployed as nonretrieved satellites. Figure 4.5-5 is a DMS block diagram for three diagnostic packages flown on Flight 1. A similar DMS block diagram for Flight 2 is shown in Figure 4.5-6. Key features of the AMPS DMS are:

- o The Spacelab CDMS provides all digital data and command requirements for AMPS.
- o Dedicated AMPS equipment does not require new development and is designed to provide flexibility to assure quick change-over from flight to flight.
- o Commonality of components exist among the data and command system for the deployed packages.
- o No addition or modification is required of the Spacelab/Orbiter hardware interface.
- o Capability is provided for autonomous experiment management by the onboard crew.
- o All data recovered on the ground is in the same format as presented to the AMPS DMS.
- o All data available to the onboard crew are available to the ground personnel.

Experiment Data Bus - As noted in Figures 4.5-3 and 4.5-4, AMPS makes effective use of the services provided by the Spacelab experiment computer, data bus, and remote acquisition units. The data bus provides a bidirectional link between the instrument and the Spacelab CDMS. Its primary task is control and monitor of instrument operation and acquisition of data of approximately 100 Kbps or less. All monitor data required by the computer, onboard crew, or by the Orbiter are programmed into the RAU. For the minor constituent experiment (see Tables 4.5-1 and 4.5-2), each instrument provides one data output to the RAU to support instrument operations and monitor functions only, while the output to the high rate multiplexer contains all the data required for post-flight analysis. Figure 4.5-7 identifies channel allocation and instrument interface guidelines. The RAU multiplexes and converts analog

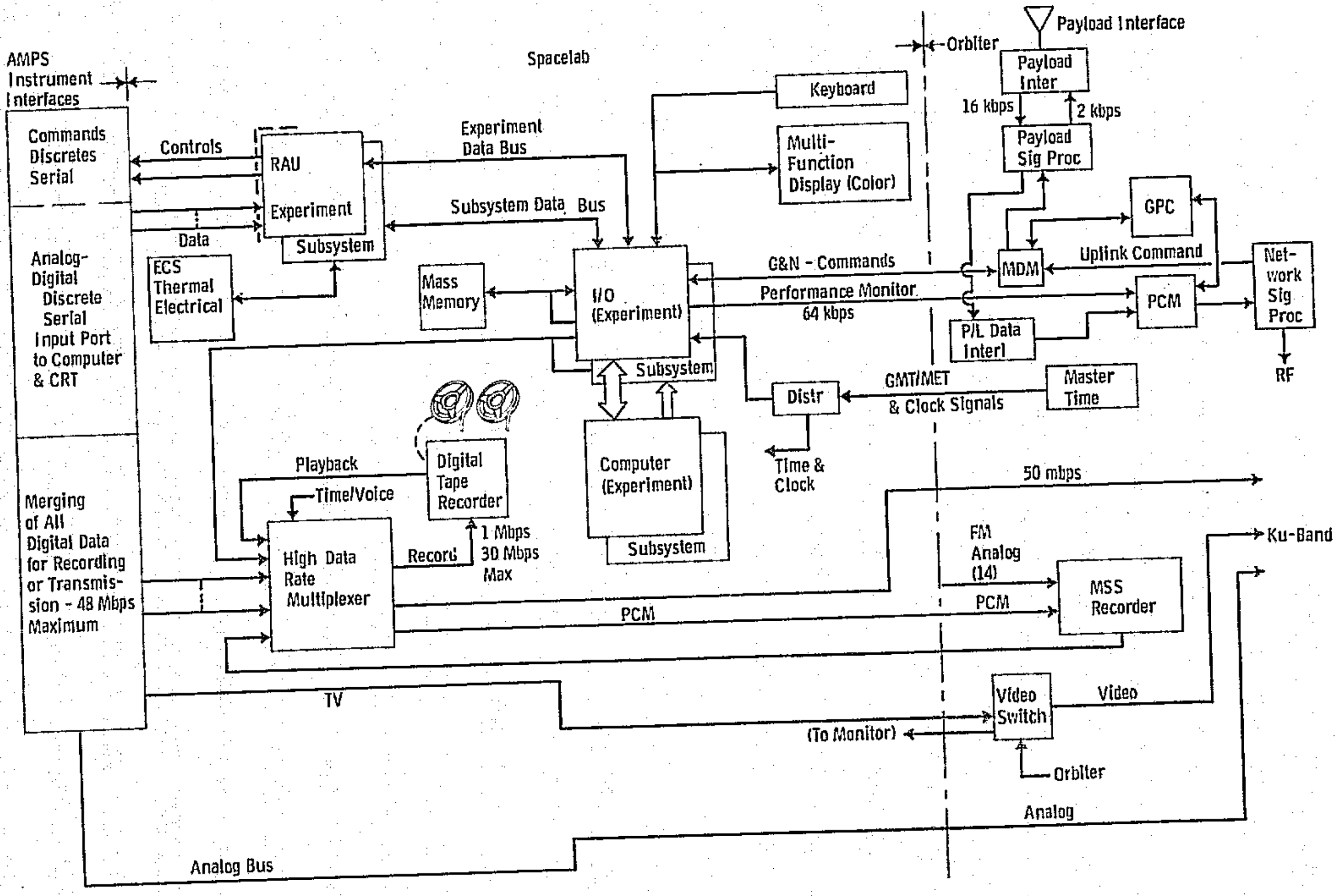


Figure 4.5-3 Spacelab-Orbiter Capability

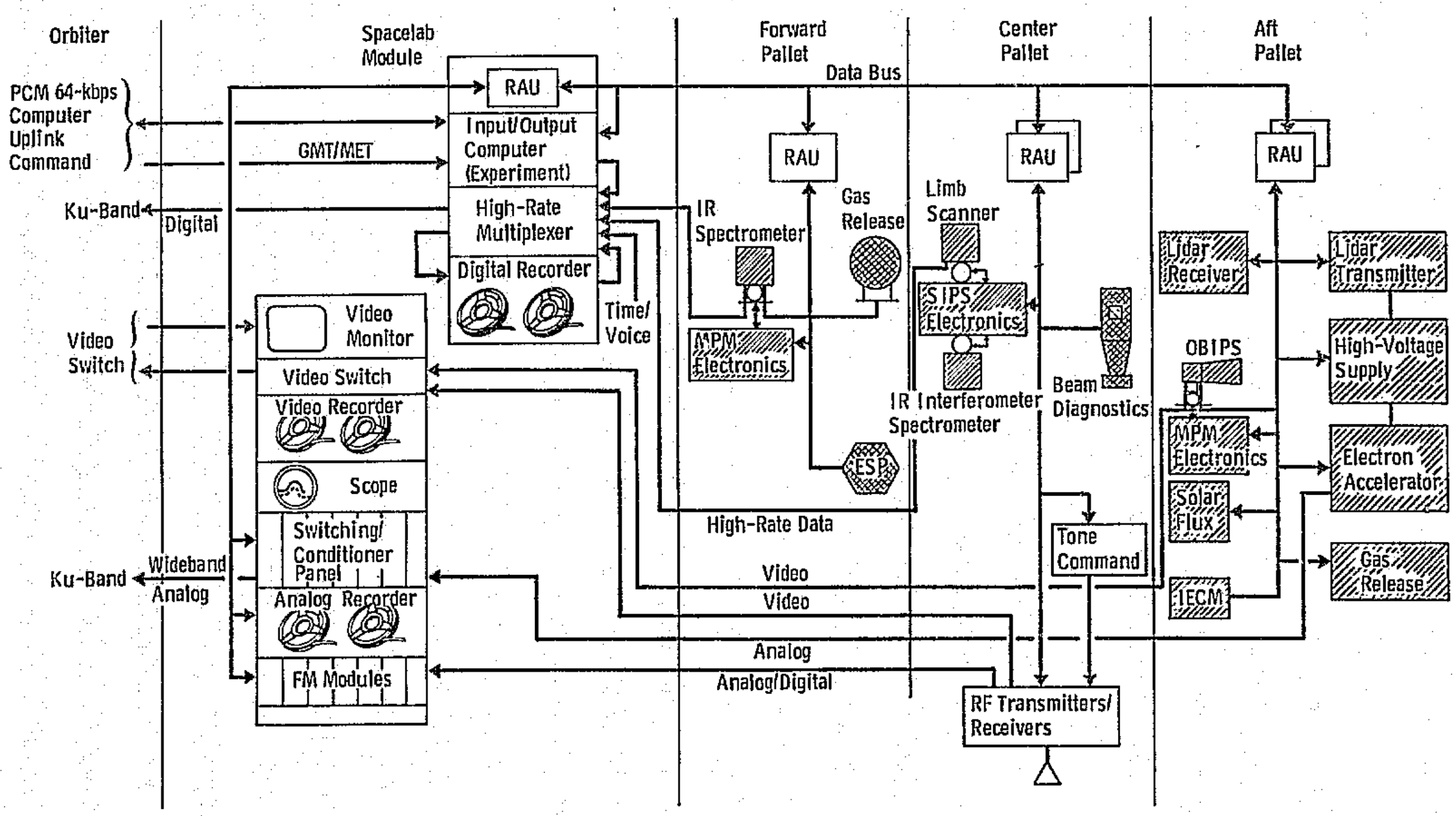


Figure 4.5-4 Baseline AMPS Data Management System - Flight 1

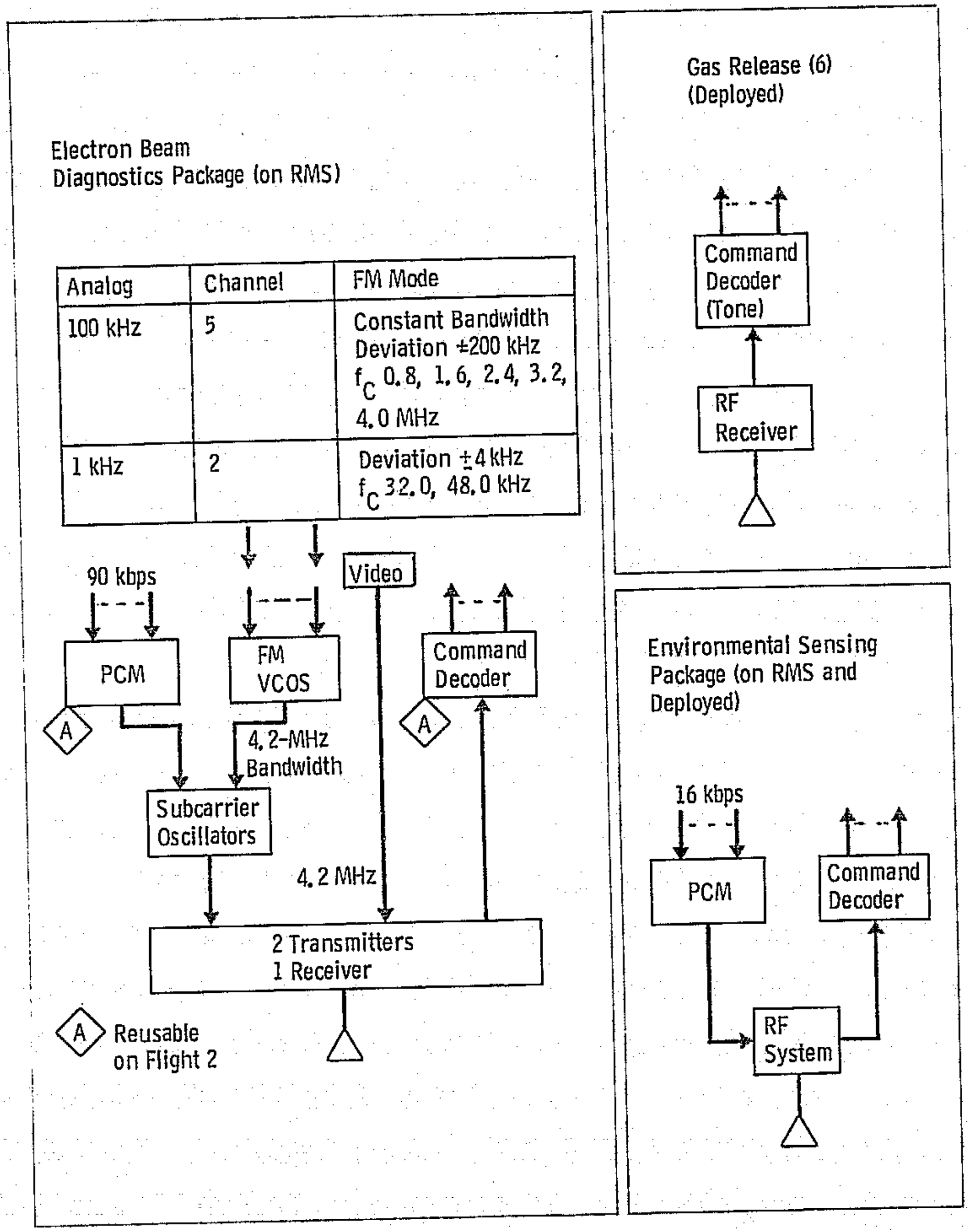


Figure 4.5-5 Flight 1 Diagnostic Packages

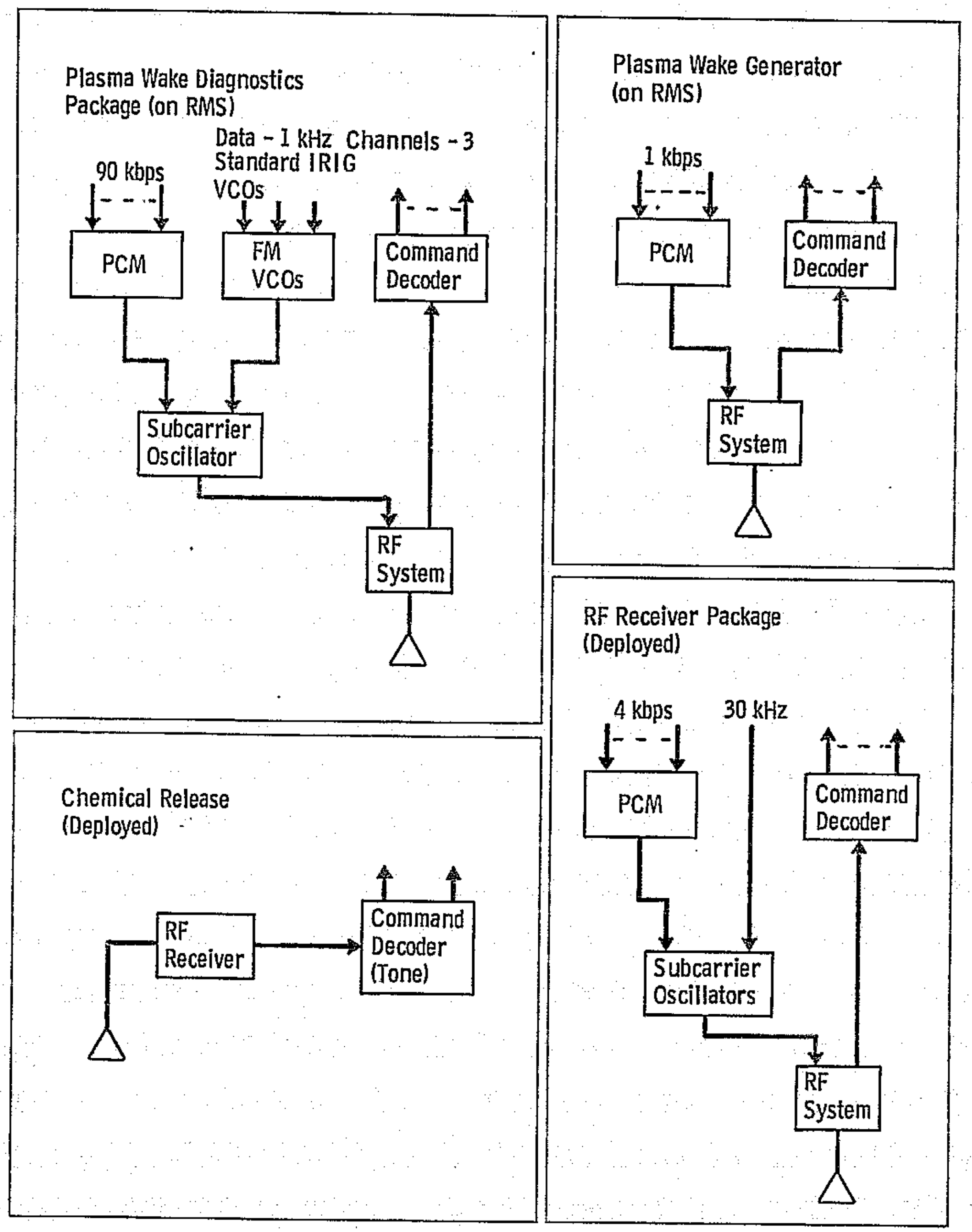


Figure 4.5-6 Flight 2 Diagnostic Packages

signals to 8 bit digital words at sample rates of 1, 10 or 100 samples per second. Serial digital data from an instrument are also transferred via a RAU in NRZ-L code and 17 bit words. Each serial channel consists of a data line, a clock line and a request line. All of the above data lines are under control of the experiment I/O and computer and performed in a demand/response manner.

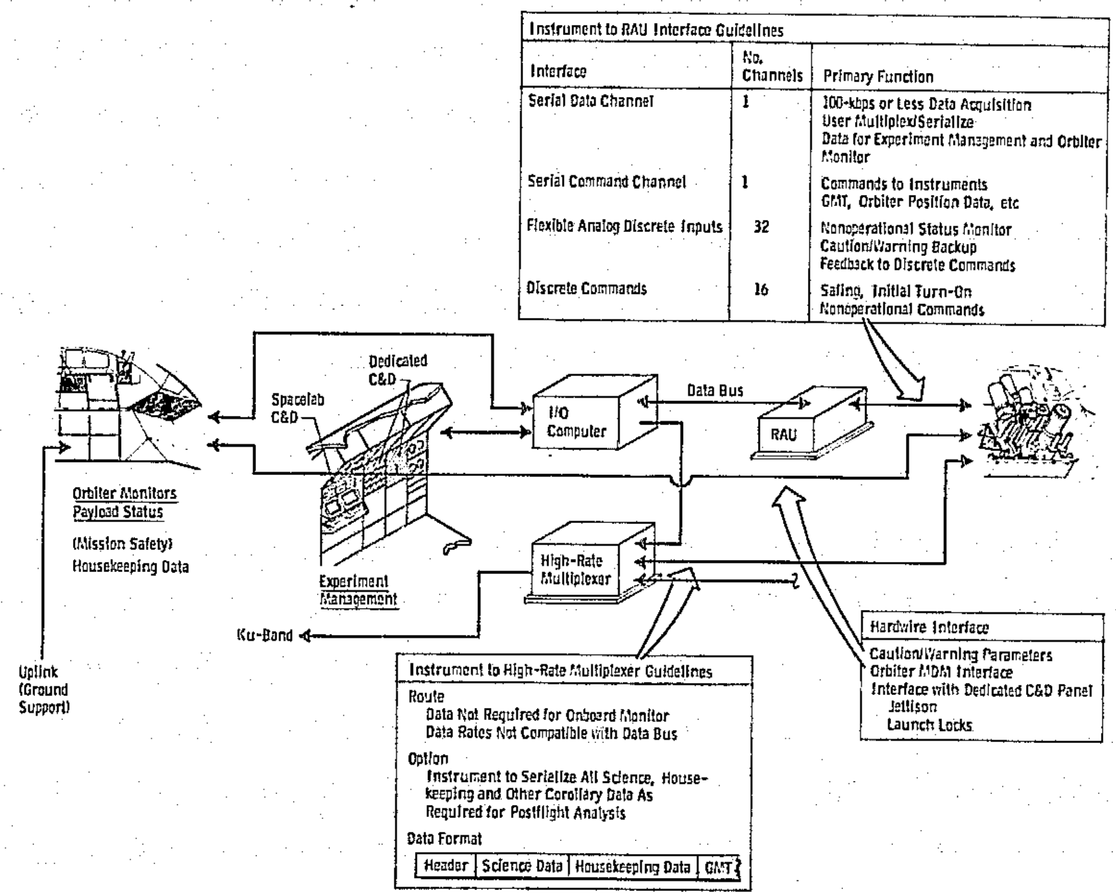


Figure 4.5-7 Pallet Instrument Interface Criteria

The experiment data bus also provides commands to the AMPS instrument and the primary command interface is the allocation of one serial command line from a RAU to each major instrument plus a limited number of discrete command outputs. The serial command interface requires a command data line and a separate clock line from the RAU to the instrument.

These data and command functions require six RAUs to meet the requirements of Table 4.5-1. Figure 4.5-4 also shows an experiment RAU inside the Spacelab module. This is required to control various recorders as well as monitor dedicated equipment performance. The prime reason for the low number of RAUs required is due to the fact that no RAUs are installed on any of the four pointing platforms. This is discussed further in Section 4.5.3.

High Rate Multiplexer/Tape Recorder - The high rate multiplexer is the central collection point of all payload and Spacelab science and corollary data. As noted in Figure 4.5-2, minor constituent experiment data between 2 and 6.3 Mbps are being continuously generated during the six day mission and are routed directly into the high rate multiplexer since the data bus cannot accommodate such data rates. It is recommended that each instrument serialize as much of their science and corollary data as possible, and route this data stream into the high rate multiplexer to facilitate post-flight data reduction.

The communication timeline studies of Section 5.8 indicate that the Spacelab provided digital recorder will be required to store data during communication gaps. With a capability to record up to 30 Mbps and provisions for replaceable tapes, the recorder does not put any constraints on the mission. Recorder utilization is discussed further in Section 4.7.

Video/Analog Support Equipment - Based on the requirements of Table 4.5-1 and the austere capability of the Spacelab Video/Analog subsystem (Figure 4.5-3), the resultant AMPS DMS includes a variety of video/analog FSE designed for quick configuration changes and capable of accommodating future growth requirements. The video recorder is provided to ensure complete coverage of video data acquisition and is of prime importance for the acoustic gravity and electron beam studies as noted in Table 4.5-1 and timelined in Figure 4.5-2. Analog recorders are used to support the electron beam study where half of the signals originate from the diagnostic packages.

Orbiter Interface and Support - The AMPS DMS design makes maximum use of the existing Orbiter interface and imposes no additional hardware requirement. All data transfer to and from the Orbiter is provided over existing wires. All communications to and from the ground are accomplished via the Orbiter communication system as described in Section 4.7. The 64 Kbps line between the Spacelab experiment I/O and the Orbiter PCM subsystem is used primarily for Orbiter monitor of the payload status and backup to the caution and warning system, with minimal usage as a mechanism to recover any science data. All science and required corollary data are transmitted to the ground via the Ku-band subsystem. Uplink commands to the AMPS payload are received by the Orbiter and transferred to the Spacelab via the Orbiter MDM. In addition the Orbiter provides, over this same line, Orbiter state vector

data (see Table 5.6.2-3) as required by the AMPS pointing system. Timing data to the AMPS payload is derived from the Orbiter Master Timing Unit. This timing signal is made available to an instrument via the experiment computer/data bus/RAU. Although the Orbiter provides data acquisition and command capability with detached payloads, this capability of the Orbiter system is not used as discussed in Section 4.5.3.

Deployed Packages - In addition to supporting DMS operations of pallet mounted instruments, Tables 4.5-1 and 4.5-2 identify the need for data/command capability for the following boom mounted or deployed packages.

(1) Flight 1

Gas Release Package	Deployed
Beam Diagnostic Package	RMS Mounted
Environmental Sensing Package	RMS Mounted and Deployed

(2) Flight 2

Chemical Release Package	Deployed
RF Receiver Package	Deployed
Plasma Wake Generator	RMS Mounted
Plasma Wake Diagnostic	RMS Mounted

The basic requirements of data acquisition and command decoding and distribution are met by use of PCM encoders, FM voltage controlled oscillators and command decoders as shown in Figures 4.5-5, 4.5-6 and 4.5-8. All digital requirements are met by using a PCM encoder of common design, with programmable bit rates, formats and standard modules to accommodate variations in input channels. FM modules are provided to multiplex analog signals. A constant bandwidth FM voltage controlled oscillator assembly is provided for the Electron Beam Diagnostic Package of Flight 1. This output along with the digital data is routed to a subcarrier oscillator assembly where each data set is allocated to a unique frequency spectrum. Thus, only one transmitter is required for communication with the pallet mounted receiver. Because of its wide bandwidth, video data is retrieved via a dedicated RF subsystem.

Two types of command decoders are used depending upon the number of commands required. Tone command decoders are used for the chemical and gas release packages, where command requirements are few, while a standard digital command decoder is used for the diagnostic packages.

As noted in Figure 4.5-4, dedicated AMPS transmitters and receivers are used for communicating with RMS mounted or deployed packages. All commands are stored in the experiment computer and are routed to the transmitter via a RAU. Tone command encoders are under computer/RAU control. Received data are routed either to a RAU (if the data is

digital PCM only) or to the FM module in the Spacelab if the data is multiplexed PCM and analog. Multiplexed data are routed to FM discriminators and the separated digital data is routed to a RAU while the multiplexed analog data is discriminated for onboard display. A detail diagram of the signal routing for the received analog/digital multiplex and video signals is shown in Figure 4.5-9. The details of the signal routing and processing of analog signals are discussed further in the supporting analysis section of 5.6.1.

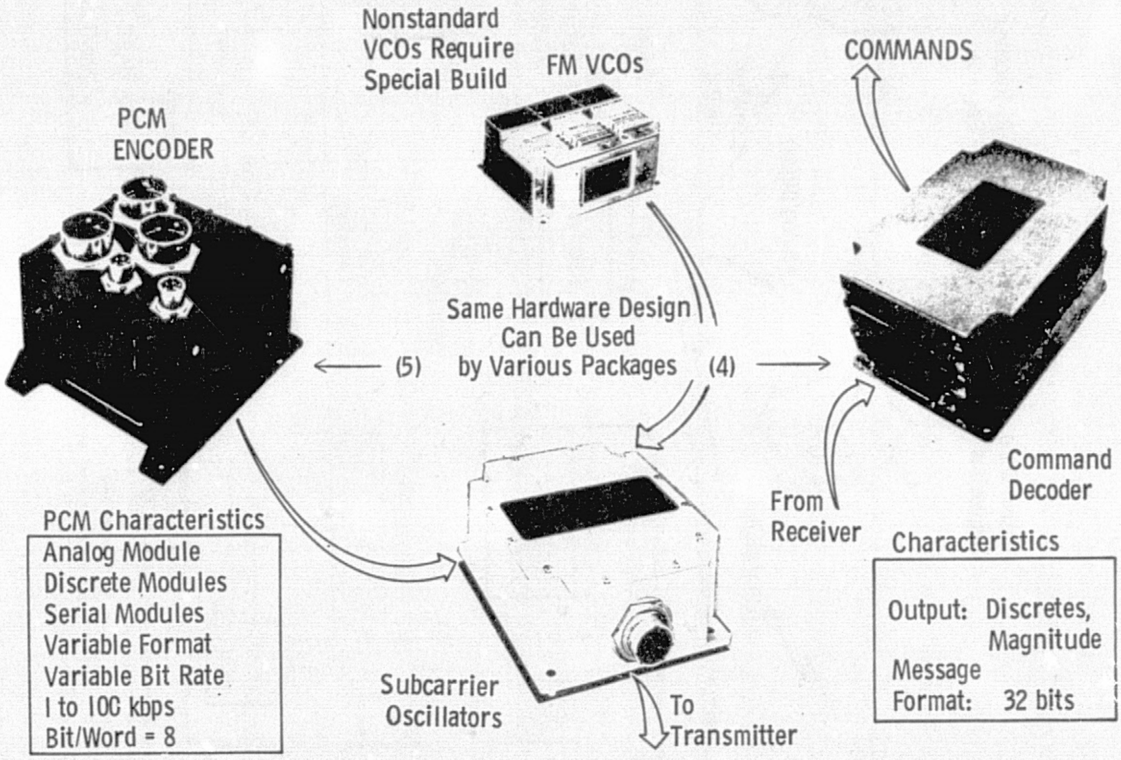


Figure 4.5-8 Common Design for RMS Mounted and Deployed Packages

Flight 2 Delta - A comparison of Flight 2 versus Flight 1 pallet mounted instrument requirements indicates reuse of ten of the science instruments/platforms flown on Flight 1, replacement of the electron accelerator by the RF plasma wave instrument on the aft pallet and the addition of the UV VIS spectrometer/photometer on the forward pallet (see Table 4.5-3). The net change has minimal impact on the baseline DMS. The UV-VIS-NIR spectrometer/photometer generates 1 Mbps of data and interfaces with the high rate multiplexer. Thus, the maximum data rate generated on Flight 2 exceeds that of Flight 1 by this amount. Modification will be required of the signal processing equipment in the Spacelab module to accommodate the needs of the deployed packages. Table 4.5-3 shows the need for two PCM encoders, three command decoders and two subcarrier oscillator assemblies to support the RMS

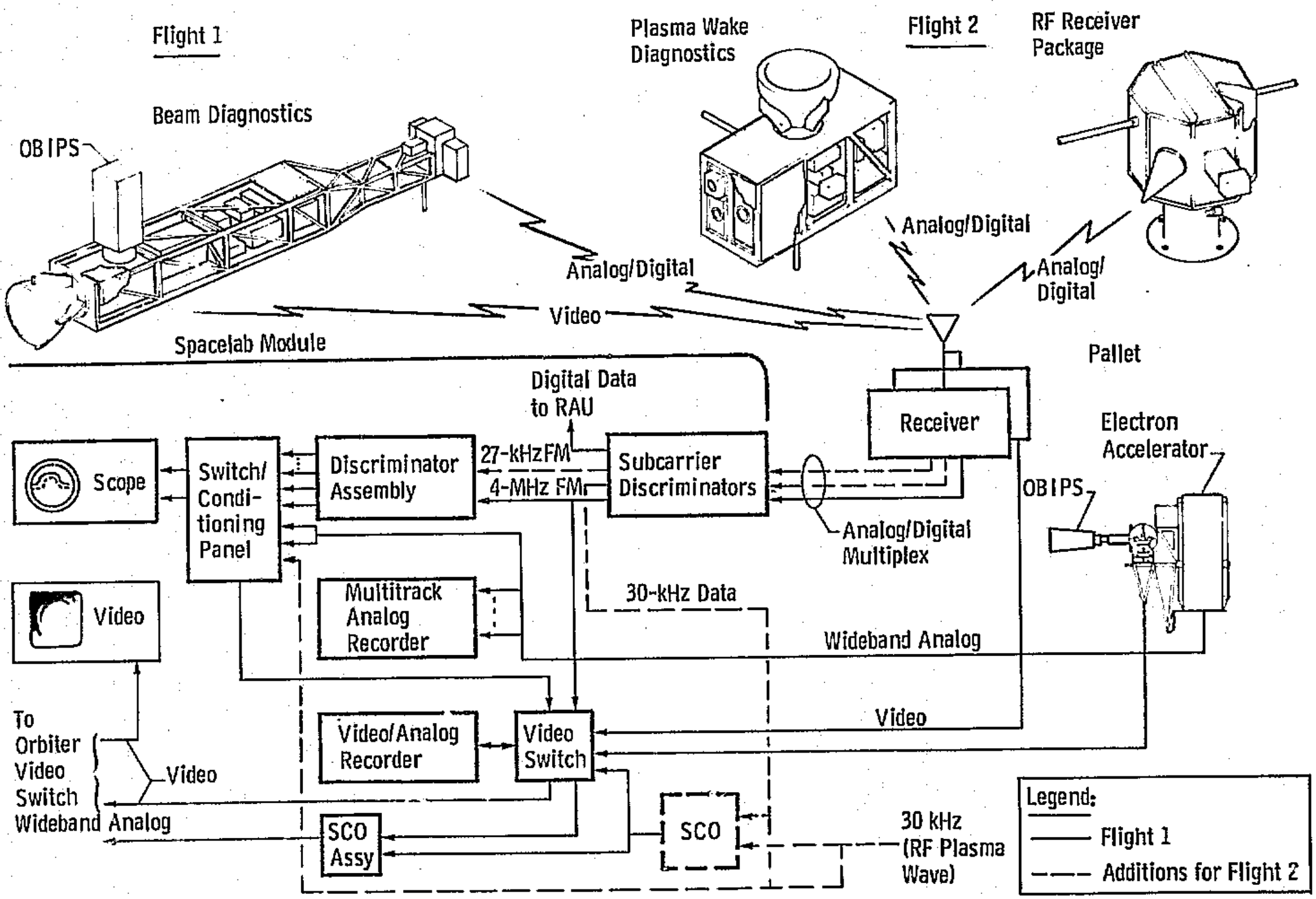


Figure 4.5-9 Video/Analog Signal Distribution

mounted or throwaway packages of Flight 2. The PCM encoder and command decoder used on the electron beam diagnostic package of Flight 1 are reused on Flight 2 for the plasma wake diagnostic package. Since only one OBIPS is used on Flight 2, there is no need to switch video signals as required on Flight 1.

Table 4.5-3 Flights 1 and 2 DMS Comparison

Flight 1	Flight 2	Delta Hardware for Flight 2
Forward Pallet		
Near IR Spectrometer	Near IR Spectrometer	-
-	UV-VIS-NIR Spectrometer/ Photometer	HiRate Multiplexer I/F
MPM	MPM	-
Gas Release (6) (Throwaway)	Chemical Release (Throwaway)	Command Decoder
Env. Sensing Package (RMS mounted & throwaway)	Plasma Wake Generator (RMS mounted)	PCM encoder/command decoder
Center Pallet		
Cryo Limb Scanner	Cryo Limb Scanner	-
Cryo IR Interferometer Spectrometer	Cryo IR Interferometer Spectrometer	-
SIPS	SIPS	-
Beam Diagnostic Package (RMS Mounted)	Plasma Wake Diagnostic Package (RMS mounted)	Subcarrier oscillator package/Discriminators (use PCM encoder & command decoder from Beam Diagnostic Package)
Aft Pallet		
OBIPS	OBIPS	-
MPM	MPM	-
Lidar Receiver	Lidar Receiver	-
Laser Transmitter	Laser Transmitter	-
Electron Accelerator	RF Plasma Wave	FM Modules
Solar Monitor	Solar Monitor	-
IECM	RF Receiver Package (Throwaway)	PCM encoder, Command decoder, subcarrier oscillators/discriminators
	NOTE _____ line indicates new instruments from flight 1	

4.5.3 Configuration Rationale

Overall Concept - The AMPS DMS is based on the concept of maximum use of Shuttle capability and providing the ground science data in the same form as it was received by the DMS, thereby providing unrestricted post-flight data processing. The basic design of the Orbiter/TDRSS provides the capability for wide band data transmission (i.e., 30 Mbps digital

and 4 MHz video). Coupled with the capability of the Spacelab high rate multiplexer/demultiplexer design that provides for all demultiplexed data to be in the same form as received by the high rate multiplexer, the total Spacelab/Orbiter/TDRSS communication and data management system lends itself to the retrieval of payload data with minimal onboard compression or processing.

RAU Allocation - RAU allocations were a function of command and data rates, instrument location, pointing platform and deployed package interface design. The primary rationale influencing RAU-instrument interfaces were as follows:

(1) Simplification of RAU-to-Instrument Interface - To keep the RAU-to-instrument interface simple, it was necessary to establish command and data line allocation as described in Figure 4.5-7. By providing at least one pair of serial data and command lines to a user, the number of wire interfaces were minimized, documentation was reduced and interfacing with an instrument micro-processor was possible.

(2) Data Input to RAU - With the availability of the high rate multiplexer for data acquisition, the RAU-instrument interface is used primarily for instrument control and monitor. Therefore, it is desirable that each instrument develop the subject interface whereby only that data required for onboard/ground monitor is routed to the RAU/computer and all instrument science and corollary data be serialized for input to the high rate multiplexer. This is particularly true of the minor constituent experiments.

(3) No Inter-pallet RAU Signal/Command Wiring - The criteria of no wiring to an RAU located on one pallet from an instrument mounted on a different pallet, minimizes inter-pallet wiring, but more importantly does not require inter-connecting these wires with an RAU simulator during a single pallet checkout.

Pointing Platform/RAU Interface - Evaluation of the RAU interfaces with those instruments mounted on the pointing platforms indicated a potential need for more RAUs than provided by the baseline Spacelab CDMS. (Eight RAUs are provided in the baseline Spacelab CDMS.) On Flight 1, two MPM and one SIPS platforms are required as schematically indicated in Figure 4.5-4. Associated with each platform is a rate gyro assembly and in the case of one SIPS platform a star tracker is also required. An analysis was conducted to determine whether an RAU should be installed on each platform or whether command and data lines should be routed through the various gimbals to RAUs located on the pallet. To assist in the evaluation, the NASA standard rate gyro and star tracker were used to identify interface requirements.

The primary consideration of the RAU-to-platform mounted instrument interface is the effect of the cabling torques on the various gimbals. The design criteria is then to minimize the cabling across the gimbals. The wire interface at the instrument/star tracker/rate gyro output for RAUs located on the platform is given by Figure 4.5-10 and for RAUs located on the pallet by Figure 4.5-11. The rate gyro data outputs are split in two, one data set going directly to the platform electronics and the other set being housekeeping data which is routed to the Spacelab experiment data bus. In this configuration, a total of nine RAUs are required for Flight 1, four for the platforms, one each in the Spacelab and the forward and center pallets and two on the aft pallet. This is one greater than the number of RAUs provided by the Spacelab. In addition, a sensor interface box is provided at each platform to condition and multiplex signal outputs required by the platform computer. Parametric comparisons of the two approaches are tabulated in Table 4.5-4.

This data indicates that locating the RAU on the platform provides less hardware impact on instrument and sensor interfaces. With a pallet mounted RAU, all rate gyro and star tracker data are routed through a larger sensor interface box and throughput to the pallet RAU via the SIPS electronics. However, (unless the RAUs are modularized), effective use is not made of the RAUs in that their channel capacity greatly exceeds the instrument/sensor requirements. The number of wires crossing the gimbals are essentially the same for either condition and the effects of the wire torques on pointing accuracies are minimal. For the baseline AMPS configuration, the biggest driver in locating the RAUs on the pallet is the cost incurred for one additional RAU and the potential loss of RAUs should a platform be ejected if its retraction cannot be accomplished.

Data Management/Controls and Display Interface - This portion of the data management system was designed with primary consideration for signal routing capabilities and provisions for quick changeover from flight to flight. For Flight 1, twelve analog signals must be available for display on the oscilloscope. Mission provided equipment include the tape recorders, FM modules and transient recorders as noted in Figure 4.5-4. Dedicated analog recorders were provided only after rejecting the use of the orbiter MSS recorder due to its limited operational capabilities. Based on discussions with JSC, the capabilities of the MSS recorder for recording analog signals are very limited. While the recorder can record analog data on the first tape pass, analog signals cannot be recorded on subsequent passes. In addition, no cabling for the analog channels are provided in the Spacelab to the Orbiter interface.

Modularity of analog FM modules is required to provide quick changeover to Flight 2. Other flights may include panels or equipment provided by the instrument developers themselves. The ability to

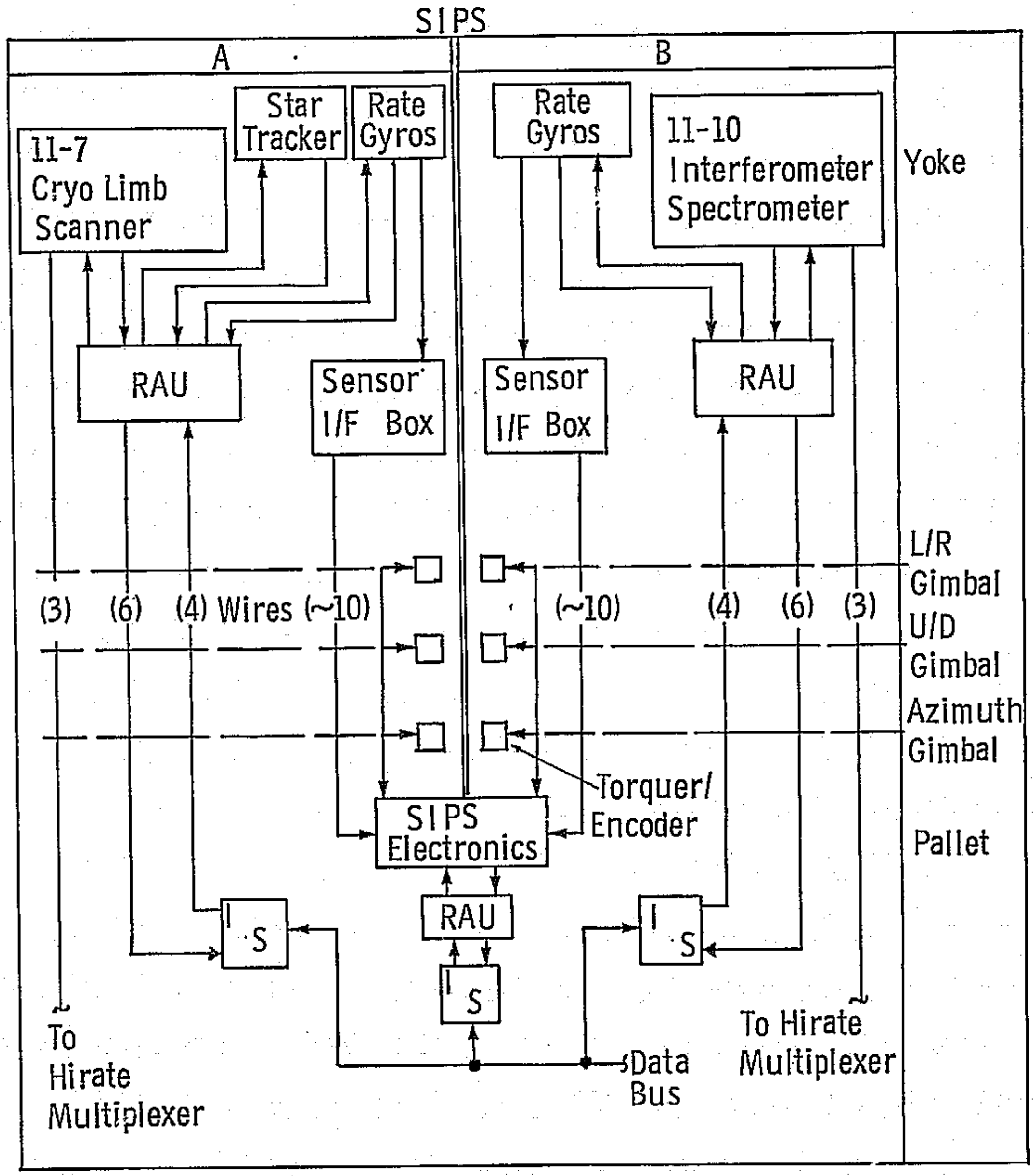


Figure 4.5-10 RAU on Canister Configuration

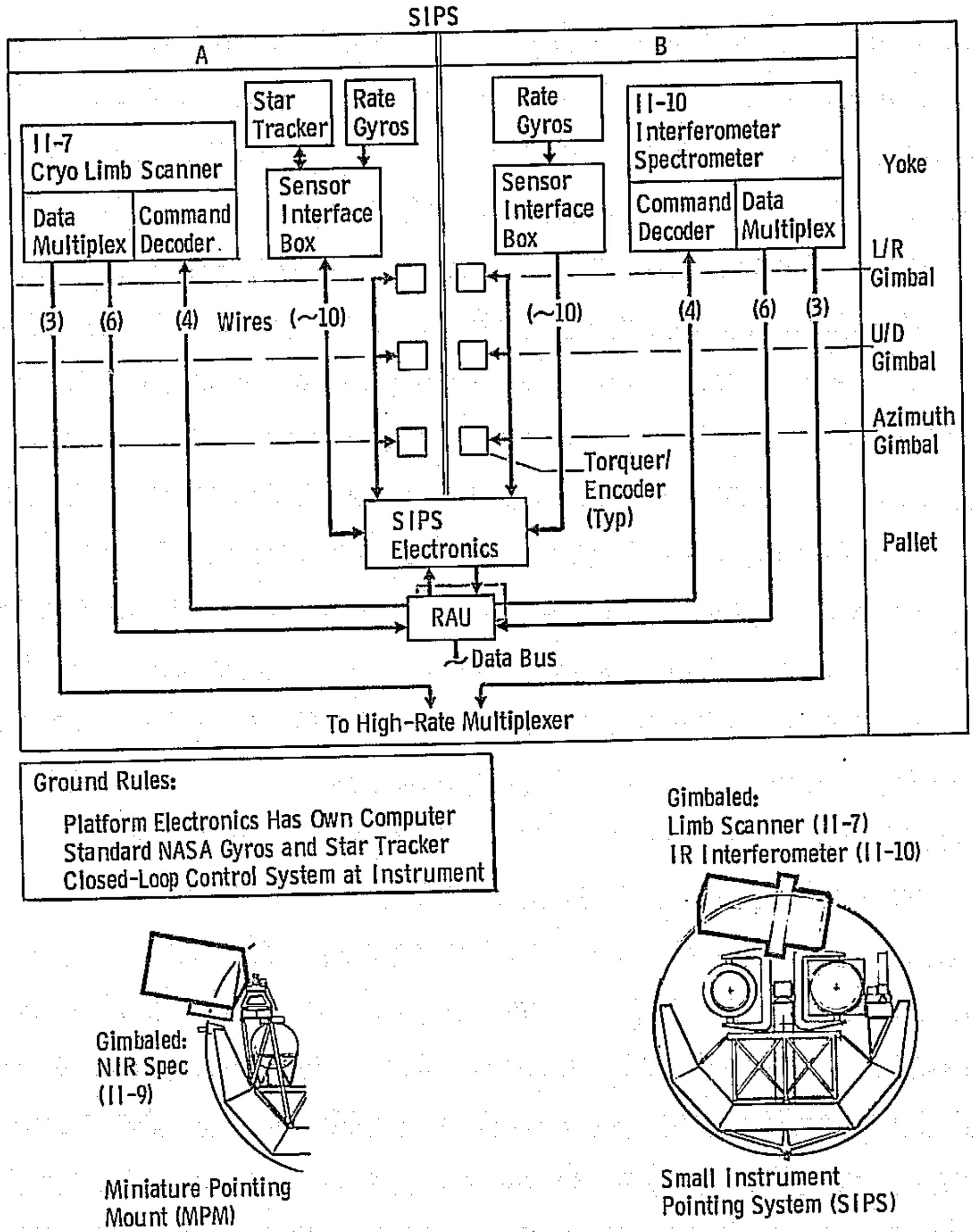


Figure 4.5-11 MPM or SIPS to Spacelab CDMS Interface

easily interface with an RAU is also provided as detailed in Section 5.6.1. For example, the digital output of a transient pulse expander is routed to the RAU input panel. RAU commands from this same panel are routed to the various recorders. A fall-out of the flexibility provided by this equipment is its potential usage among non-AMPS payloads. This subsystem can be enhanced even further by the application of the NIM/CAMAC modules also discussed in Section 5.6.1.

Table 4.5-4 RAU Allocation Comparison

Parameter	RAU on each Platform	RAU on Pallet
Total No. RAU	9	6
Instrument Interface	RAU provides flexible command and data interface. Minimal instrument multiplex/format required.	Instrument must multiplex more signals. Larger multiplexer required.
Rate Gyro Interface	Housekeeping and command signals via RAU. All other data to Sensor Interface box.	All data must be routed through sensor interface box.
Star Tracker Interface	All signals via RAU	All signal transfer via sensor interface box.
Number of Wires Across Gimbal	23 - 28	23 - 28
Emergency Ejection of Platform	Loss of atleast one RAU; two for SIPS	No loss of RAU

Deployed Packages Interface - As noted in Figure 4.5-12, data and command interfaces with deployed packages are via AMPS provided transmitters/receivers mounted on the pallet. This decision was made as a result of 1) reviewing the Orbiter S-band capabilities for subsatellite communications, identifying its short comings for AMPS and 2) the trade study whereby an RF system instead of a hardwire system was selected for the RMS deployed packages.

The Orbiter communication system does have command and data acquisition capabilities with subsatellites as shown in Figure 4.5-12. While the Orbiter command rates are compatible with the AMPS deployed package requirements, the data rates generated by certain diagnostic packages were incompatible with the Orbiter capability as noted in Table 4.5-5. Since the Beam Diagnostic Package on Flight 1 and the RF Receiver Package and Plasma Flow Diagnostic Packages of Flight 2 produced data rates far in excess of the Orbiter capability, the need for dedicated RF equipment became obvious. Secondly, use of the Orbiter system would have required routing of received data from the Orbiter to the Spacelab I/O where subsatellite data must be continuously monitored. This interface as shown in Figure 4.5-12 does not exist and would require modification to both Spacelab and Orbiter hardware.

Table 4.5-5 Detached Payload Versus Orbiter Capability

Diagnostic Package	Digital Data	Analog Data	Video Data	Orbiter Capability
Flight 1				
Gas Release	0.1 kbps			Orbiter detached
Beam Diagnostic Package	97 kbps	100 KHz (5) 1 KHz (2)	4.2 MHz	payload interface
Environmental Sensing Package	16 kbps			can receive digital data up to a
Flight 2				maximum of 16 kbps.
RF Receiver Package	4 kbps	30 KHz		
Plasma Flow Test Body	1 kbps			
Plasma Flow Diagnostic Package	90 kbps	1 KHz (3)		

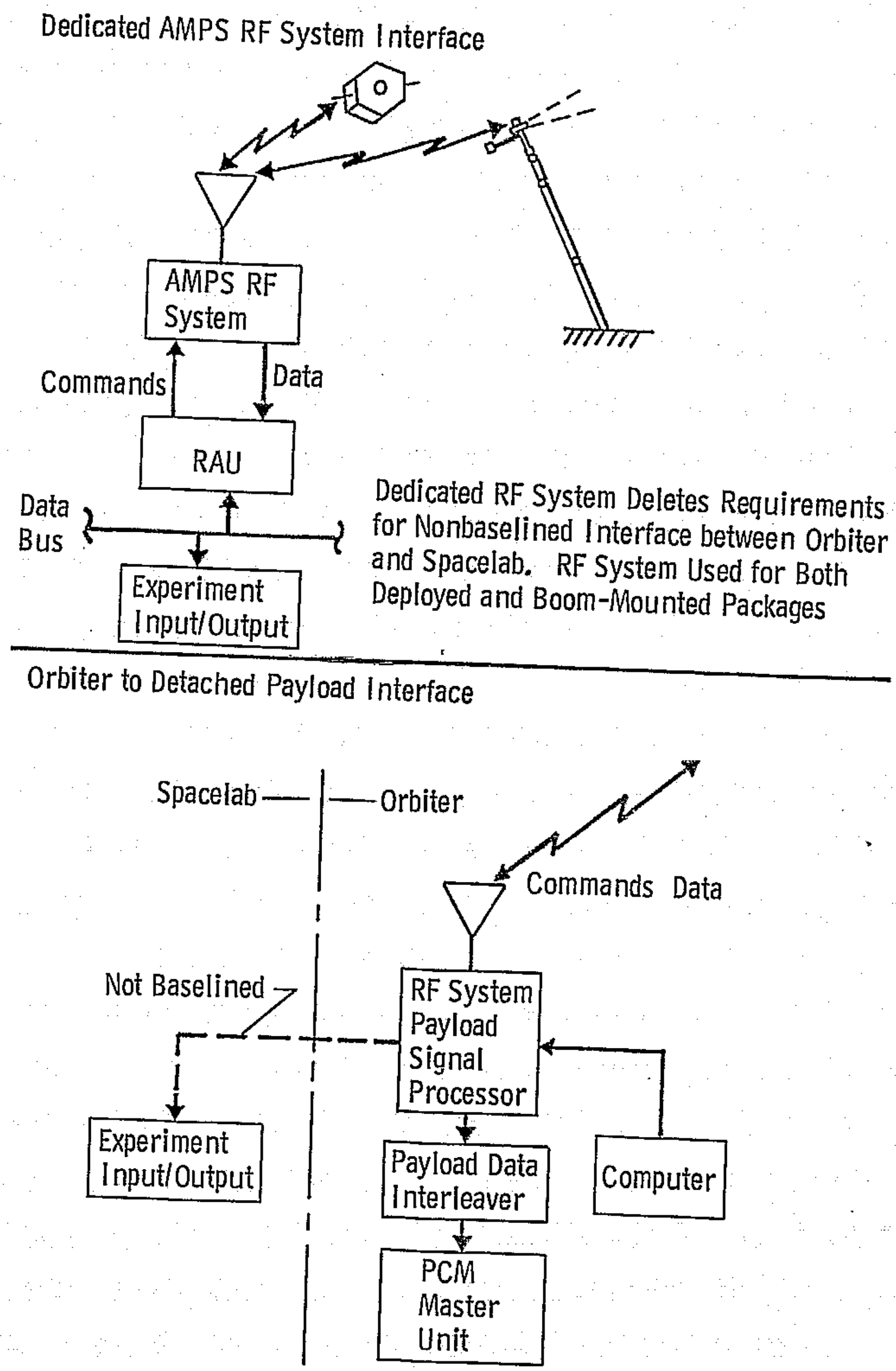


Figure 4.5-12 Detached Payload Options

All digital data requirements of Table 4.5-5 for deployed packages are met by using a PCM encoder of generic design. Modularity in the encoder provides capability to meet the varying requirements of input channel, bit rate, data length, subcommutation and supercommutation. This provides a cost effective approach with respect to logistics, checkout and maintenance. The use of a common PCM encoder design also results in a common telemetry format. The format of the minor frame can be the same for all of the diagnostic packages. Typical format characteristics for the Electron Beam Diagnostic Package are shown in Section 5.6.2. It is characterized by a minor frame length of 250 words and 8 bit word lengths resulting in a data rate of 100 kbps. Other data rates are generated by programming a bit rate divisor in the basic oscillator circuitry.

The decision to provide two types of command decoders was based on the fact that each of the six gas release packages require a small number of commands and the packages are not retrieved. The use of tone command decoders provides a low cost approach capable of accommodating the required number of commands. This same design is used for the chemical release package of Flight 2. A digital command decoder of standard design is used for the other diagnostic packages. A modular design provides discrete or digital outputs to meet each package requirements.

Table 4.5-6 shows that the beam diagnostic package imposes the biggest demand on analog signal processing. The design approach was to provide a constant bandwidth FM approach limiting the total bandwidth of the multiplexed signal to 4.2 MHz, the capability of the Orbiter Ku-band system. Using a guard band between channels equal to at least two times the deviation, the following channel allocations were made resulting in a modulation index of 2 for the high frequency channel and 4 for the low frequency channels.

Table 4.5-6 Constant Bandwidth Channel Characteristics

CHANNEL	DATA BAND WIDTH	CENTER FREQUENCY	DEVIATION
1	100 kHz	4.0 MHz	±200 kHz
2	100 kHz	3.2 MHz	
3	100 kHz	2.4 MHz	
4	100 kHz	1.6 MHz	
5	100 kHz	0.8 MHz	
6	1 kHz	.048 MHz	± 4 kHz
7	1 kHz	.032 MHz	

<u>Conclusion</u> - The AMPS DMS provides a system capable of supporting Flights 1 and 2 with minimal modification between flights. Effective use is made of the Spacelab/Orbiter/TDRSS capability for data acquisition supplemented by data tapes stored onboard. Dedicated equipment will be required for each diagnostic package with commonality of components or design wherever practical. Since the Spacelab has no capability to process analog data, AMPS provided FM modules, transient recorders and video/analog recorders are provided. This assembly is designed for flexible signal routing which provides ease of reconfiguration and potential use for other payloads. It is recommended that the need for such high cost items such as the video recorder be reviewed across a broad spectrum of Spacelab payloads to determine whether it should be included in the category of Labcraft type equipment. This could result in significant cost savings to the payload projects.

4.6 Control and Display Subsystem

This section describes the definition of the AMPS Control and Display Subsystem. It includes the establishment of the C&D functional and operational requirements, the concept definition and rationale for the configuration.

4.6.1 Control and Display Requirements

The C&D functional requirements were developed for each instrument and then expanded to provide the detailed functions necessary to define the hardware, software, and man/computer interface requirements. These functional requirements provided a data base from which individual flight requirements were extracted as the payloads were defined. The Flight Support Equipment (FSE) functional requirements were developed as the configurations became available.

Tables 4.6-1 and -2 summarize the C&D functional requirements for AMPS Flight 1 and 2 respectively. The tables define the quantities and types of controls and displays (D = 2 or 3 variable discretes; D(m) = more than 3 variable discretes; and A = analog/high resolution functions) and graphics required for each instrument and FSE subsystem. This data provided inputs to communication, data management, software analyses in addition to C&D analyses. The detailed functional requirements are tabulated in Appendix F.

Table 4.6-1 Flight 1 C&D Functional Requirements Summary

Instrument/FSE		Commands			Displays			Graphics
		D	D(m)	A	D	D(m)	A	
I-1	Laser Sounder	19	2	12	19	2	10	3 Plots
I-9	Electron Accelerator	3	4	4	3	4	3	3 Oscilloscope
I-21	Chemical Release (6)	24	6	6	24	6	6	
II-3	OBIPS (2)	12	2	(6)	12	2	0	2 Video (CCTV)
II-7	Cryo Limb Scanner	3	0	5	3	0	12	
II-9	NIR Spectrometer	3	0	4	3	0	11	
II-10	IR Spectrometer	3	0	5	3	0	12	
III-2	Vector Mag	2	2	0	2	2	0	
III-3	Level I Diag	2	0	2	2	0	3	
III-4	Level II Diag	5	0	4	5	0	1	7 Oscilloscope
III-16	Ion Mass Spect	3	1	1	3	1	1	
III-18	RPA	2	2	3	2	2	3	2 Plots
III-23	Neutral Mass Spec	3	4	1	3	4	4	4 Plots
III-22	Langmuir Probe	2	1	1	2	1	6	1 Plot
III-25	ESP - includes AC/DC field only; must include III-2, III-22, III-16 for full ESP requirements	2	0	0	2	0	2	2 Plots
	Gimbal System (3)	20	4	12	20	4	12	
	HV Power Supply	1	1	2	1	1	7	
	Release Mech (7)	42	7	0	42	7	0	
	Stowage Mech (2)	4	0	0	4	0	0	
	TOTAL	$\underbrace{155 \quad 36}_{526}$		68	$\underbrace{155 \quad 36}_{526}$		93	12 Plots 10 Oscilloscope 2 Video

Table 4.6-2 Flight 2 C&D Functional Requirements Summary

	Instrument	Commands D	Commands D(m)	Commands A	Displays D	Displays D(m)	Displays A	Graphics
I-1	Laser Sounder	19	2	12	19	2	10	3 Plots
I-12	RF Plasma Wave (Fixed)	5	6	5	5	6	7	4 Plots
	RF Plasma Wave (RMS)	2	2	3	2	2	5	4 Plots
I-21	Chemical Release (1 Unit)	4	1	1	4	1	1	
II-3	OBIPS (1)	6	1	(3)	6	1	0	1 Video-CCTV
II-4	UV-VIS-IR Spect (2 Units)	8	8	6	10	8	4	2 Plots
II-7	Cryo Limb Scanner	3	0	5	3	0	12	
II-9	Near IR Spect	3	0	4	3	0	11	
II-10	IR Spectrometer	3	0	5	3	0	12	
III-2	Vector Mag. (3)	6	6	0	6	0	9	
III-10	Ion Mass & Dist Analy	3	1	1				
III-17	Deployable Test Body	2	1	1				
III-18	Planar RPA	2	2	3	2	2	3	2 Plots
III-22	Langmuir Probe	2	1	1	2	1	6	1 Plot
III-23	Neutral Mass Spect	3	4	1	3	4	4	4 Plots
	Gimbal Platforms (Assume 3)	20	4	12	20	4	12	
	Release Mech (2)	12	2	0	12	2	0	
	Stowage Mech (2)	4	0	0	4	0	0	
	TOTAL	$\underbrace{112 \quad 35}_{434}$		61	$\underbrace{104 \quad 33}_{406}$		96	

The significant payload flight operations requirements used in the analysis are listed below. These requirements resulted from analyses of mission and payload operations and generally apply to all payloads. They comprise planning and scheduling functions, experiment/FSE operations and data analysis, and payload status monitoring and corrective action.

- o Experiment planning/scheduling updates
- o Coordinate ground/flight operations
- o Experiment powerup, checkout, calibration
- o Initiate/terminate experiment operation
- o Real-time data monitoring and quick look analysis
- o Data processing for PI analysis
- o Update target pointing data
- o Instrument orientation/fine pointing
- o Experiment parameters optimization
- o Preparation, verification, and implementation of experiment program modifications
- o Monitor payload operations/status
- o Control for deployable systems
- o Fault isolation and analysis

4.6.2 Control and Display Concepts

The Spacelab and Orbiter have extensive capability to support payload C&D subsystems. The most significant capabilities are listed below.

(1) Spacelab Command and Data Management Subsystem (CDMS)

Computer Interactive C&D Subsystem/CRT/Keyboard (2)
Standardized Data Bus Interface
Dedicated C&D Interface With CDMS
Ground Command Uplink

(2) Spacelab Standard Equipment Racks

Six 19-in. Racks
Connector Mounting Plates
Power Switching Panel
Thermal Control - Forced Air

(3) Orbiter Aft Crew Station

Spacelab Data Bus Interface/CRT/Keyboard
Closed Circuit Television (CCTV)
RMS C&D

The Spacelab CRT/keyboard, shown in Figure 4.6-1, located at the Orbiter aft flight deck and the two CRT/keyboards in the module are the principal experimenter/payload interfaces. The CRTs are tricolor systems with 12 inch screens capable of displaying vectors, graphics, and alphanumeric data (21 lines, 47 characters per line). A standard alphanumeric keyboard with full ASCII capability and a 25 key function keyboard with functions selected by the user are the principal command input devices. The user is required to supply the operational software. Dedicated payload C&D, mounted in standard 19 inch racks, can be hardwired to the payload or can interface with the Spacelab data bus via the Remote Acquisition Units. Ground payload command capability is provided via the Orbiter communication system and Spacelab data bus. The two crew stations available to the payload user are shown in Figure 4.6-2.

Our approach to the AMPS C&D subsystem design, resulting from a trade study described in Section 5.7.1 makes maximum use of available Spacelab and Orbiter capability. Figure 4.6-3 is a functional schematic of the C&D subsystem for Flight 1. The majority of AMPS operations are performed in the Spacelab module. The principal experimenter/payload interface is the two S/L CRT keyboard systems. OBIPS video data is displayed on an AMPS pro-

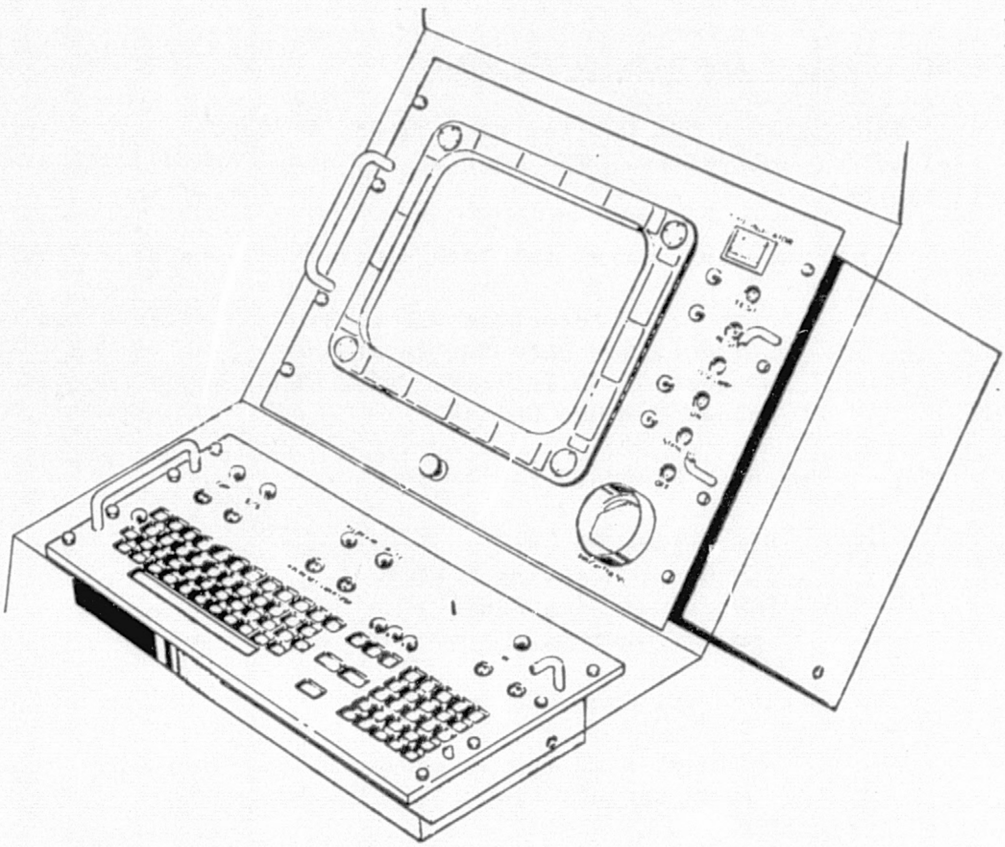


Figure 4.6-1 Spacelab CRT/Keyboard

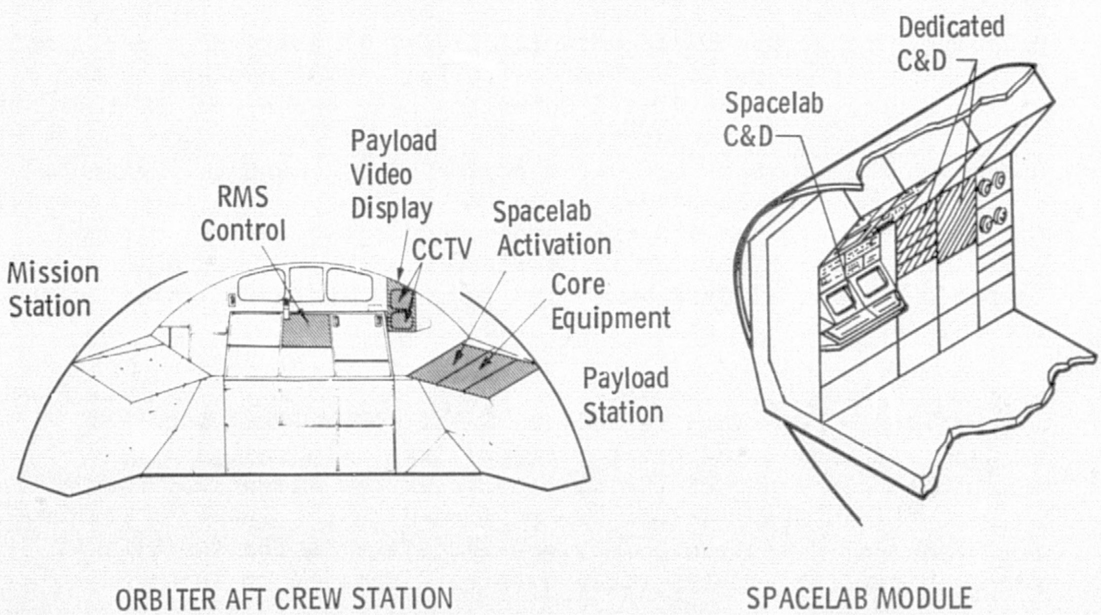


Figure 4.6-2 AMPS Crew Station Layouts

vided TV monitor located in the Spacelab module and also on the Orbiter CCTV monitors at the Orbiter Aft Flight Deck during Remote Manipulator System operations. A dedicated time display panel provides GMT, orbit phase data, and two general purpose event timers. Wide-band data from the Electron Accelerator Experiment and Level II Beam Diagnostics Experiment are displayed on an oscilloscope. A hardwired dedicated safing panel provides redundancy with the data bus for critical safety functions. The requirement for this panel may dimish when more detailed reliability and safety analyses are performed. A dedicated panel is also required to control OBIPS time critical analog functions during the Gas/Chemical Release Experiments.

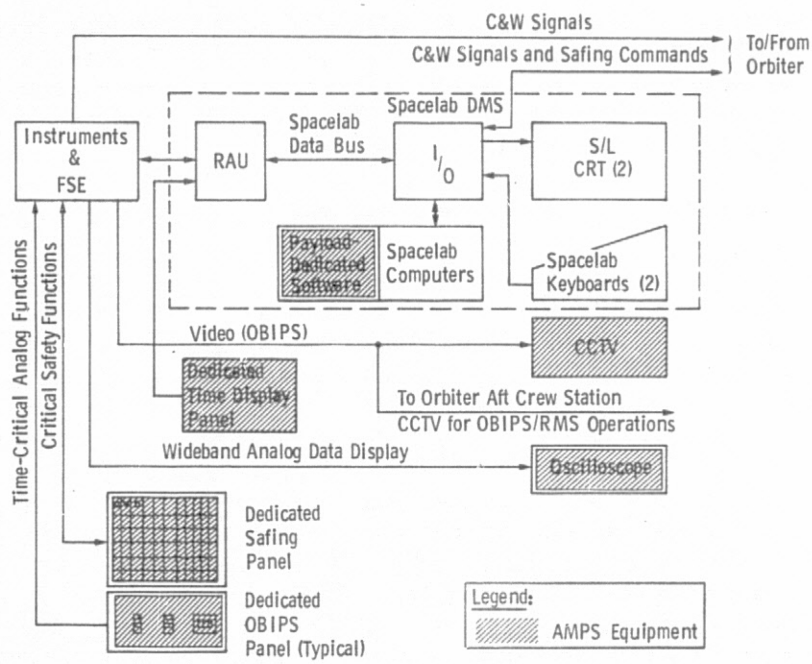


Figure 4.6-3 Flight 1 Control and Display Subsystem

The Spacelab Caution and Warning (C&W) system is integrated into the Orbiter C&W system. The Spacelab user is required to supply C&W signals, generated and conditioned to Orbiter specifications, to the Orbiter system. Signal level detection is performed in the Orbiter

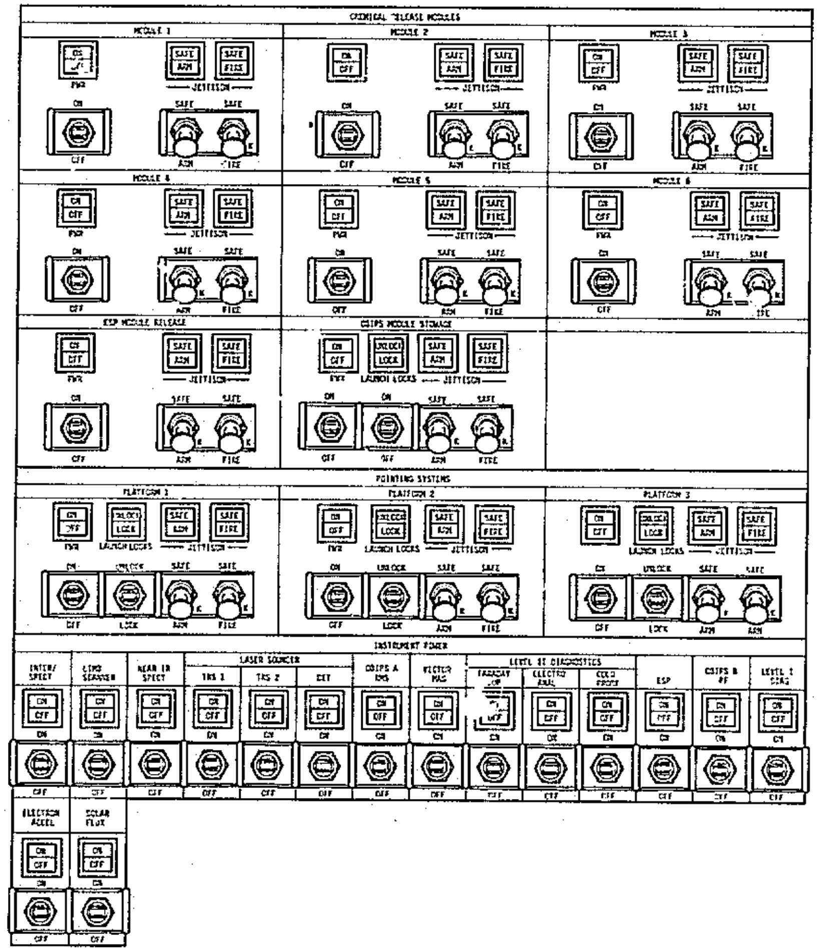
General Purpose Computer (GPC) and in the Spacelab Subsystem Computer for redundancy. The C&W signals are hardwired to an Orbiter MDM. A redundant path is provided by the Spacelab Data Bus (RAU, Subsystem Computer) and the Orbiter PCM Master Unit. Hardwired inputs to the Orbiter C&W Electronics Assembly are also provided. C&W conditions are displayed on the Orbiter and Spacelab CRTs. The Orbiter GPC will drive the Master Alarm and intercom alarm signals in the Spacelab Module. The safing commands are initiated by Orbiter keyboard entry to the GPC from the forward crew station. The payload physical C&W interface is the Spacelab Forward Endcone Feedthrough.

Dedicated C&D layouts, shown in Figure 4.6-4, were prepared in order to identify interface requirements. No significant impact on Spacelab interface support capability was identified.

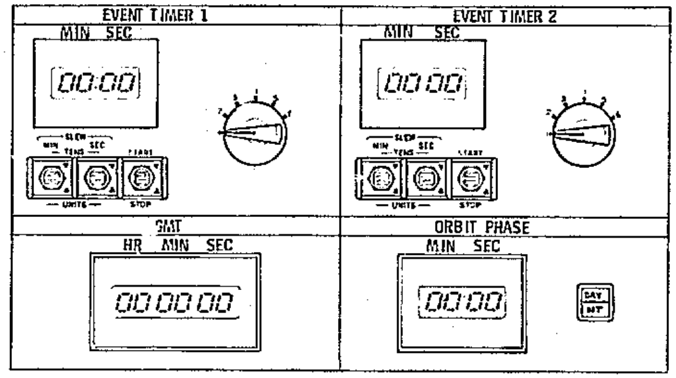
Figure 4.6-5 shows typical interface requirements for the oscilloscope and TV monitor. The payload requirements for these items must be further defined before actual hardware can be selected. The TV monitor characteristics pertain to the 525 line Orbiter CCTV which is a candidate for use by AMPS.

The overall approach to payload operations has a significant impact on the configuration of the C&D subsystem. The allocation of payload operational functions between the Payload Specialist and the ground support groups depends on their relative attributes. Table 4.6-3 defines our assessment of these attributes. The Payload Specialist has a real-time continuous interface with the payload which allows him to effectively interact with the payload and efficiently optimize performance. He is not constrained by the communication coverage limitations defined in Section 5.8.1. The principal attributes of the ground are the extensive manpower and computational facilities capable of supporting flight activities in the areas of performance monitoring, data reduction and analysis, and detailed activity planning and coordination. Table 4.6-4 shows a preliminary allocation of functions defined in Section 4.6.1, based on these relative attributes, for the AMPS payloads. The Payload Specialist has been assigned functions relating to efficient experiment performance optimization and safety. The ground has been assigned functions corresponding to their attributes.

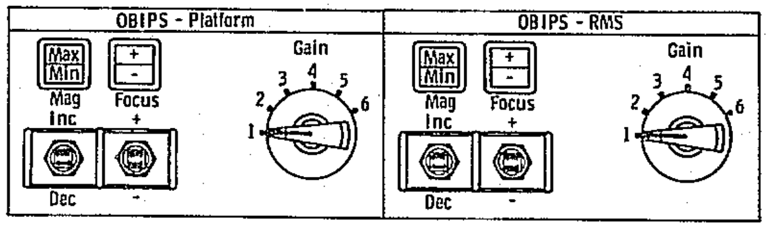
Another area which has been addressed is the allocation of functions between manual and automatic operations. The results of the task analysis, described in Section 5.7.2, and subsequent simulation activities indicated that, from an experimenter/computer interface standpoint, operations should be automated to the maximum extent consistent with the relative capabilities of the experimenter and the operating system. Experimenter functions should relate to his unique capability for judgment such as experiment data analysis and optimization. The system should be allocated to those functions which it can perform more efficiently and accurately such as experiment



Size 483 x 840 mm
Weight 18 kg
Power 32 W
Feedthroughs 240



Size 483 x 240 mm
Weight 8.4 kg
Power 22 W



Size 483 x 110 mm
Weight 3 kg
Power 2.5 W

Figure 4.6-4 AMPS Dedicated C&D Hardware Interface Requirements

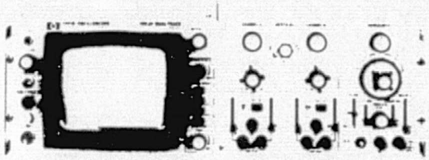
setup and configuration, limit checking, and other standardized type of operations.

Oscilloscope

Size: 483x133 mm

Weight: 20 kg

Power: 115 W



TV Monitor

Compatible with EIA Standards 170/33

Orbiter CCTV Characteristics

Size: 483x178 mm

Weight: < 50 kg

Power: 35 W

Figure 4.6-5 Oscilloscope/TV Monitor Interface Requirements

Table 4.6-3 Payload Specialist/Ground Attributes

Payload Specialist Attributes	Ground Attributes
Real Time Payload Interface a) Experiment Performance Optimization - Quick Look Data Analysis - Parameter Optimization - Real Time Planning b) Reaction to short time constant phenomena c) Efficient performance of iterative analog functions Selective Data Acquisition a) Offload Ground Processing Requirements b) Offload Communication/Onboard Data Storage Requirements	Manpower a) Hardware Specialists - Routine Data Monitoring - Fault Isolation/Analysis b) Scientists - Data Analysis - Experiment Program Mods c) Operations Specialists - Mission Operations Planning - Remote Facility Coordination Computational Facilities a) Large Capacity b) Unlimited Manpower c) Cheaper to Develop

Table 4.6-4 Operations Allocation

Crew Functions	Ground Functions
Powerup, Checkout, Calibration	Data Processing for PI Analysis
Initiate/Terminate Operations	Experiment Planning/Scheduling Updates
Data Monitoring/Quick Look Analysis	Preparation, Verification, and Implementation of Experiment Program Modifications
Instrument/FSE Parameter Optimization	Update Target Pointing Data
Instrument Pointing/Orientation	Coordinate Ground Operations
Control of Deployable Systems	Monitor Payload Operations/Status
	Fault Isolation/Analysis

Figure 4.6-6 shows our approach to the AMPS experiment operation and includes the allocation of functions between the Payload Specialist, the ground, and the operating system. The Payload Specialist selects the experiment from the flight plan. The system will select and configure the instruments and FSE corresponding to the selected experiment. Target selection and acquisition is a shared function which depends on the specific experiment. The Payload Specialist then verifies readiness, initiates operation, performs real-time quick-look data analysis, and optimizes experiment parameters as required. The ground performs detailed data reduction and analysis and updates the daily flight plan in consultation with the Payload Specialist.

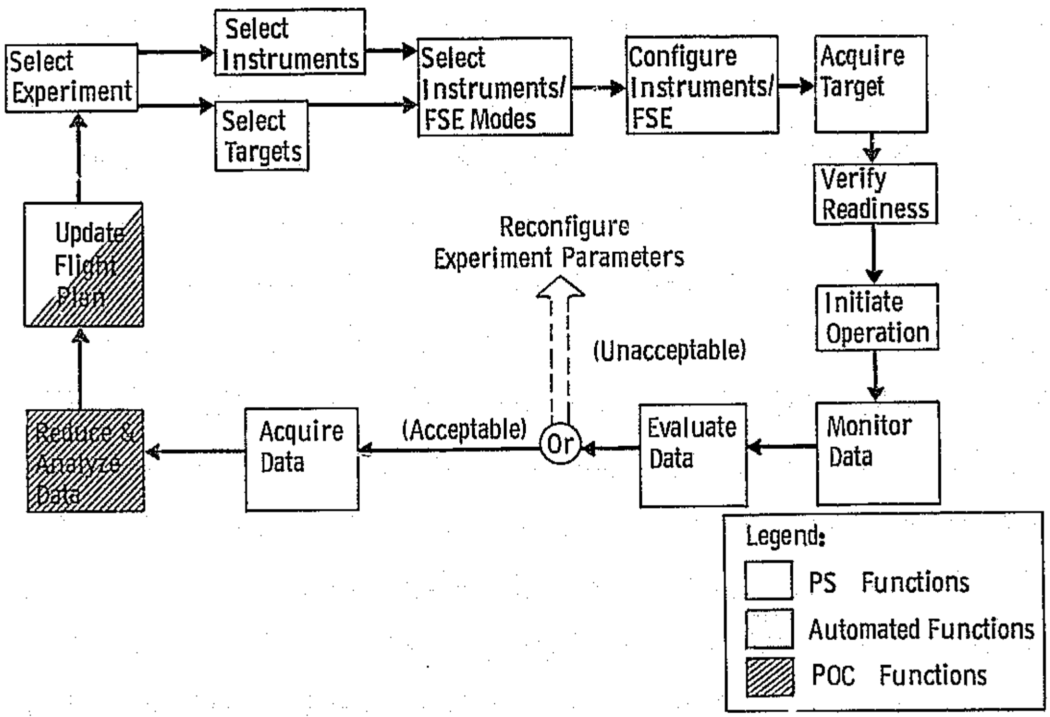


Figure 4.6-6 Experiment Operation Implementation

A simulation was developed as an engineering tool for use during the experimenter/computer dialogue development activities. The objectives of the simulation were to evaluate various types of dialogues, investigate the implementation of the function keyboard, and identify the capabilities, limitations, and constraints imposed on AMPS operations by the Spacelab hardware configuration. A typical Laser Sounder experiment was used as the subject of the simulation. The procedure included the selection, setup, calibration, and operation of the experiment. The software hierarchy is shown in Figure 4.6-7.

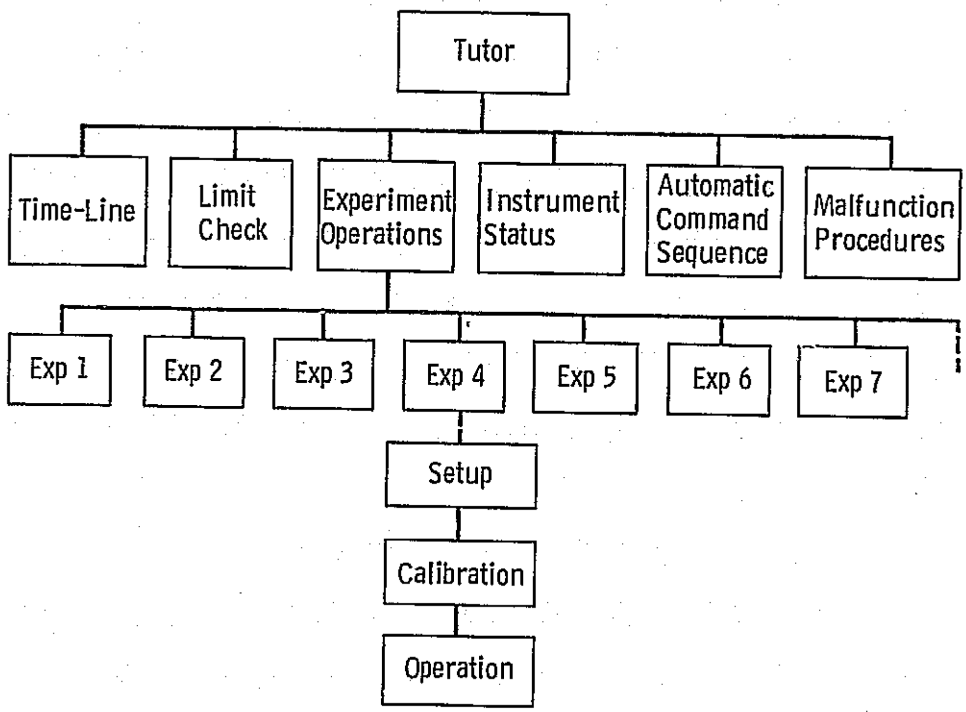


Figure 4.6-7 Experimenter Operations Control Hierarchy

The simulation is described in detail in Section 5.7.3. The results of the simulation were:

(1) Alphanumeric keyboard command inputs are undesirable since they are time consuming and have a high error probability.

(2) The menu selection approach (multiple choice) is a simple but effective method for inputting commands quickly and with a minimum of error.

(3) The Spacelab function keyboard is effective for inputting repetitive types of frequently used commands and is a desirable supplement to the alphanumeric keyboard.

(4) The Spacelab CRT has a restricted information display capability and will require highly optimized display formats in order to meet complex payload operational requirements.

4.7 Communication Subsystem

The communication subsystem described in this section will include a discussion of the Orbiter subsystem utilization as well as the definition of dedicated AMPS hardware required to satisfy an extensive array of communication requirements. Since communications imply the transfer of data between two or more terminals, we will begin by briefly describing the overall AMPS communication system with the aid of Figure 4.7-1. Major communication terminals include the Orbiter, the Tracking and Data Relay Satellite System (TDRSS), a domestic satellite (DOMSAT) relay, and the Spaceflight Tracking and Data Network (STDN). The primary routes of data transfer of interest to AMPS include:

- o Air-to-ground data exchange from the Orbiter and via the TDRSS;
- o Orbiter/AMPS to deployed packages;
- o TDRSS terminal to GSFC via DOMSAT.

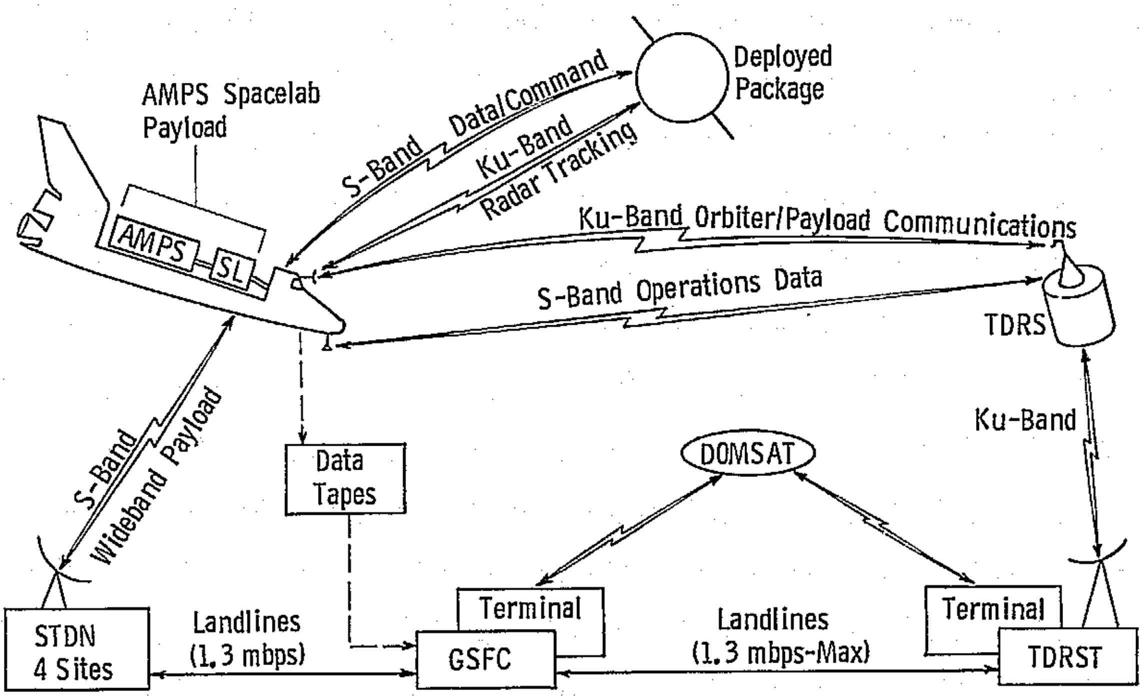


Figure 4.7-1 AMPS Communication System

The primary link for the transfer of AMPS data is the Orbiter-TDRSS Ku link whose capabilities are summarized in Figure 4.7-2. This RF link can be operated in two communications modes, one which handles up to 30 Mbps of digital data, and another which can handle a combination of digital and analog data. This Ku system can also be operated

as a radar link, and will be used to track AMPS deployed instrument packages.

Ku-Band (TDRSS)				S-Band		Detached Payloads
Mode	Channel 1	Channel 2	Channel 3	TDRSS	STDN	
1	50 Mbps*	2 Mbps	---	64 kbps Interleaved with Orbiter Data	Digital 5 Mbps or Analog 4 MHz	TM - 16 kbps Command - 8 kbps Coded
2	4 Mbps or 4.2 MHz	192 kbps Operational Data	2 Mbps			

*Digital rates in excess of 30 Mbps result in less than 3-dB margin.

Assumptions

- o Ku-Band Rendezvous Radar Can Be Used to Track AMPS Deployed Packages (JSC Confirms).
- o Nominal Landline and DOMSAT Leased Capability Will Be 1.3 Mbps.
- o Additional DOMSAT Capability Is Available for Wideband Digital and Analog Data.
- o DOMSAT Terminal Co-Located at GSFC.

Figure 4.7-2 Orbiter Communications

The Orbiter-TDRSS S-band link is used primarily for Orbiter operational data although this includes up to 64 Kbps of payload data. In addition, the Orbiter provides the capability to command and receive limited digital data from deployed packages. Back-up links to the STDN are available and can handle wideband digital or analog data.

Since the TDRS terminal (TDRST) has no capability for data storage or processing, some thought must be given to distribution of AMPS data to other terminals. A proposed distribution concept is shown in Figure 4.7-3, which assumes that the Payload Operations Control Center (POCC) is located at GSFC. The upper portion of the figure shows the individual data signals which can be generated within the payload or the Orbiter, and which can be extracted individually from the TDRSS downlink carrier and retransmitted to some ground terminal. Obviously the 192 Kbps signal containing Orbiter operational data and flight critical payload data will be transferred to the flight control center at JSC. There then exists another group of less critical data which is required to closely monitor and control payload experiment operations. Current data requirements definitely indicate that this data could easily be contained in the 2 Mbps signal; and that this signal

should be transferred via DOMSAT to the POCC for near real time processing. Additionally, the remaining wideband digital or analog/video data should be transferred to the POCC for subsequent off-line processing. This approach will limit the need for payload data processing software to only one location, and avoid costly redundant capability at different locations.

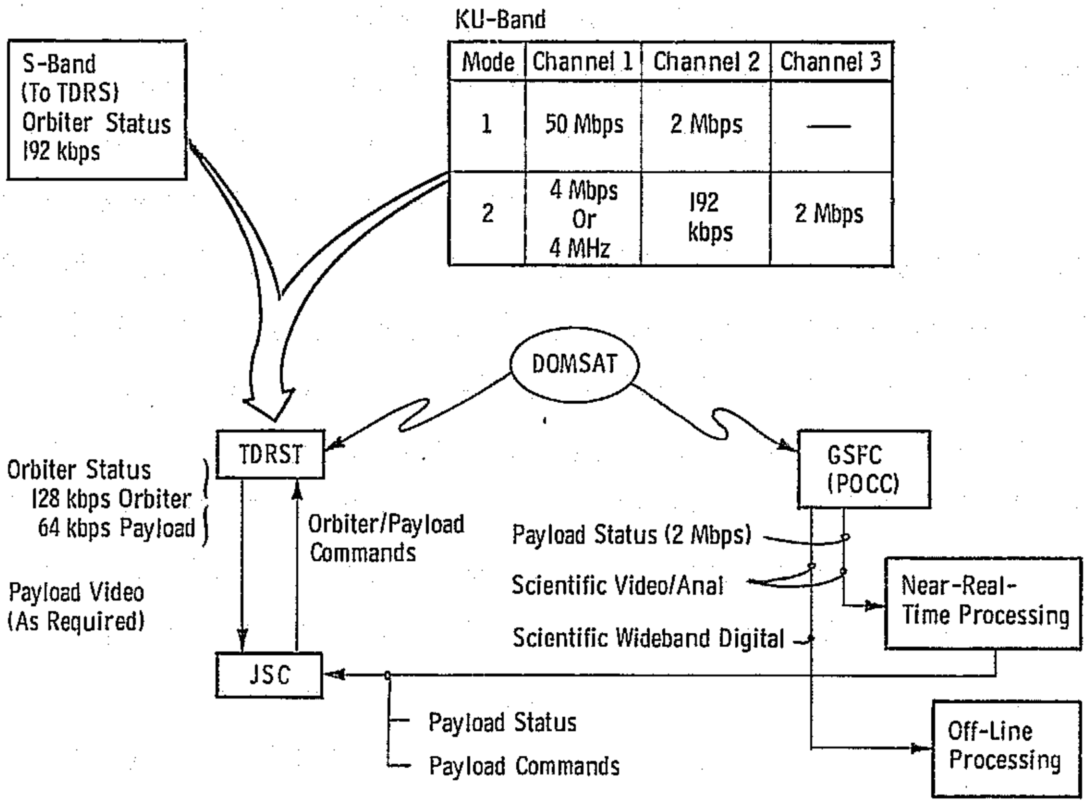


Figure 4.7-3 Ground Data Distribution

4.7.1 Communication Requirements

The AMPS functional communication requirements are summarized in Table 4.7-1. Additional quantitative and mission oriented details will be provided as these requirements are discussed further in the following sections.

Table 4.7-1 AMPS Communication Requirements

Data Terminal	Functional Requirements	
	Flight 1	Flight 2
Orbiter	Payload-to-Ground Communications, Data Retrieval and Command	
Gas Release	Tracking and Command	----
Chemical Release	----	Tracking and Command
ESP	Tracking, Command, and Data Retrieval (Digital)	----
RF Receiver	----	Tracking, Command, and Data Retrieval (Digital and Analog)
RMS Instruments	Command and Data Retrieval	

4.7.2 Flight 1 Communication Concepts

This section will describe various options that were evaluated for satisfying the above flight requirements, looking first at Orbiter communications and then considering solutions applicable to the various deployed packages. The tracking requirement associated with deployed packages as indicated in Table 4.7-1 is discussed in this section since it utilized part of the Orbiter communications system. In addition, the concepts presented for RF link design associated with the deployed packages did consider the impact of such radiation directed toward Earth and the resultant flux density levels, as described in Section 5.8.3.

Orbiter-To-Ground Communications - All Orbiter and payload data is transferred to and from the ground control centers via the Orbiter-to-TDRSS links. Because of its relatively wide bandwidth requirements, AMPS will rely primarily on the Orbiter Ku band communication link which utilizes a steerable, high-gain antenna deployed on a short boom from the forward section of the payload bay. One such antenna is Orbiter provided, and an optional second antenna is available, but chargeable weight and power-wise to the payload. Because of the close

proximity of these antennas to the Orbiter body and payload bay doors, line-of-sight from the antenna(s) to the TDRS is sometimes blocked, thus curtailing communications coverage via the Ku link. A detailed analysis of this blockage problem was performed for several applicable vehicle attitudes, and is described in Section 5.8.1. For the +Z-LV attitude predominant to Flights 1 and 2, cumulative single antenna coverage over a six day simulation was 45%, and dual antenna coverage reached 67%. Pertinent contact data is summarized in Figure 4.7-4. These data seem to indicate the need for extensive onboard recording during periods of antenna blockage. However, as indicated in Section 4.5, there are also some serious limitations in available recorder capability. The major advantages/disadvantages associated with the two extremes of maximized real time transmission versus maximum recording are pointed out in Figure 4.7-5 and result in the indicated concept for Orbiter-to-ground communications.

	Primary Antenna			Dual Antennas		
Parameter	Z-LV	+Y-LV	-Y-LV	Z-LV	+Y-LV	-Y-LV
Cumulative Coverage (per cent)	45	50	76	67	76.5	78
Maximum Contact (minutes)	38	27	85-90	85-90	85-90	85-90
Maximum Gap (minutes)	108	49	43	82	43	43

Z-LV Coverage Variations (Daily)

Antenna(s) \ Period	9 Rev (15 Hr)	5 Rev (8 Hr)	Cumulative
Primary	35%	60%	45%
Dual	57%	87%	67%

Figure 4.7.4 TDRSS Coverage

REAL-TIME DATA TRANSMISSION	RECORD DATA & RETURN TAPES
Requires Dual Ku Antennas Reduces Available Payload Weight 67% Coverage in Z-LV Additional DOMSAT Bandwidth Required DOMSAT Terminal at GSFC	. Additional Analog/Video Record Capability Tape Change Capability - Analog Weight of Extra Tapes Reduces Ground Interaction
RECOMMENDED AMPS CONCEPT Use Single Ku Antenna Real Time Data Transmission When Link is Available - Record During Gaps and Dump Few Tape Changes Required Above - Nominal DOMSAT Bandwidth	

Figure 4.7-5 Data Retrieval Options

In order to justify the concept of a single Ku antenna, several data recovery situations were examined to verify concept compatibility. The first of these situations is presented in Figure 4.7-6, which shows an eight-hour period of Minor Constituents (MC) experiment activity. The MC data represents the peak digital data rate encountered, 7.6 Mbps for real time (RT) data and 15.2 Mbps for combined RT and delayed time (DT) data. This activity is scheduled daily on both Flights 1 and 2, during a period of minimum TDRSS coverage as indicated for the primary and kit antennas on the figure. Only the primary antenna coverage is assumed for the single antenna concept. As the figure shows, the kit antenna does not improve coverage substantially during this period.

When TDRSS coverage is not available, data is recorded for periods up to 80 minutes before tape change is necessary. As the figure shows, maximizing the transmission of RT and DT data reduces the required tape changes to two during this activity. An even longer period of MC activity is presented in Figure 4.7-7, but is accompanied by a higher percentage of communications coverage. This results in decreased data recording and, with the loss of less than 30 minutes of data, no need for a tape change.

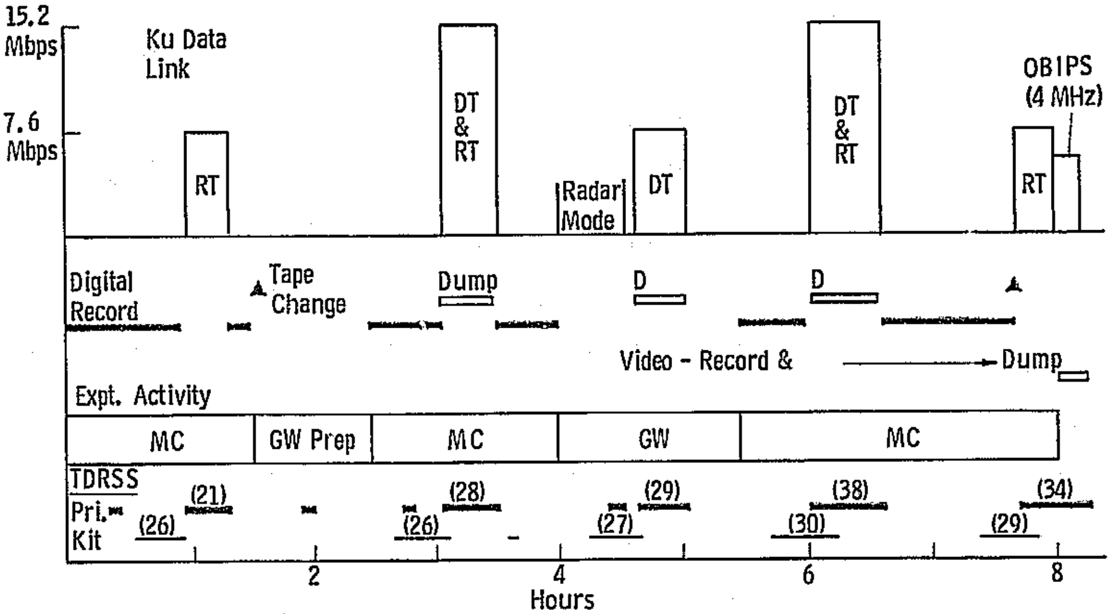


Figure 4.7-6 Data Recovery Timeline A

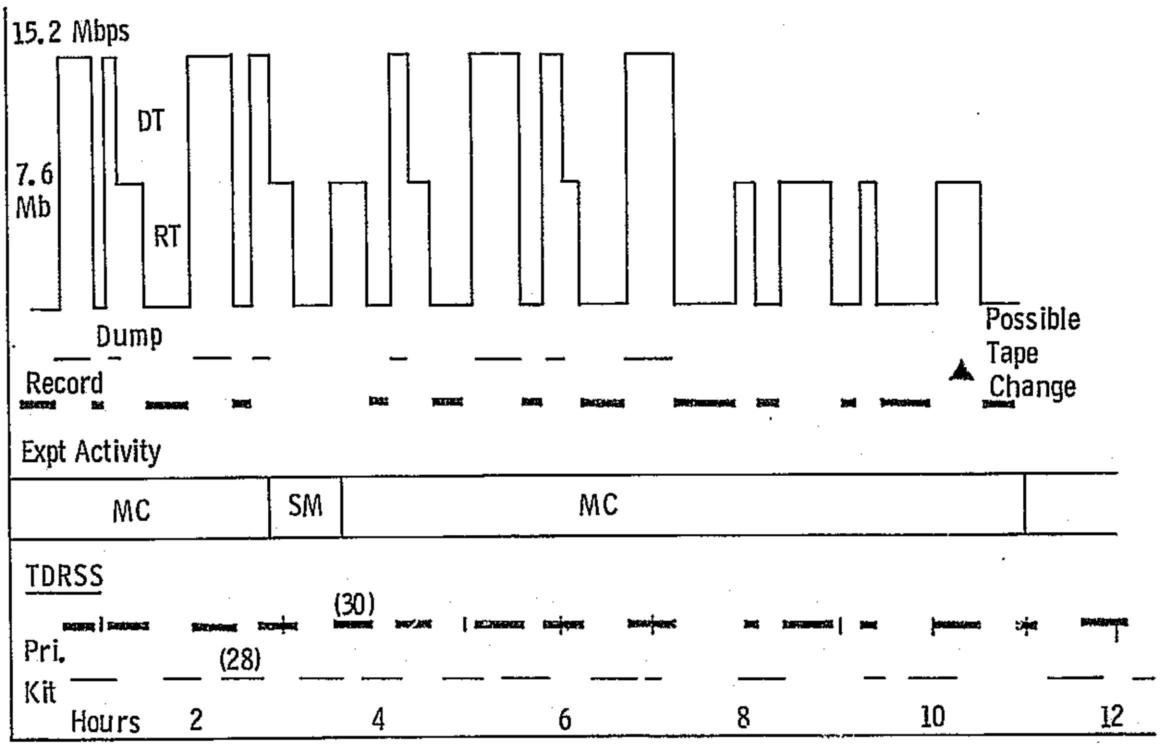


Figure 4.7-7 Data Recovery Timeline B

Figure 4.7-8 shows a period of beam diagnostics experiment activity during which wideband video or analog data is generated. This activity has intentionally been scheduled to take advantage of the higher TDRSS coverage because the analog/video recorder has a limited record time (5 minutes) and a tape change capability. It can be seen that adequate communication opportunities are available to recover RT data as well as to clear the recorder before the tape runs out.

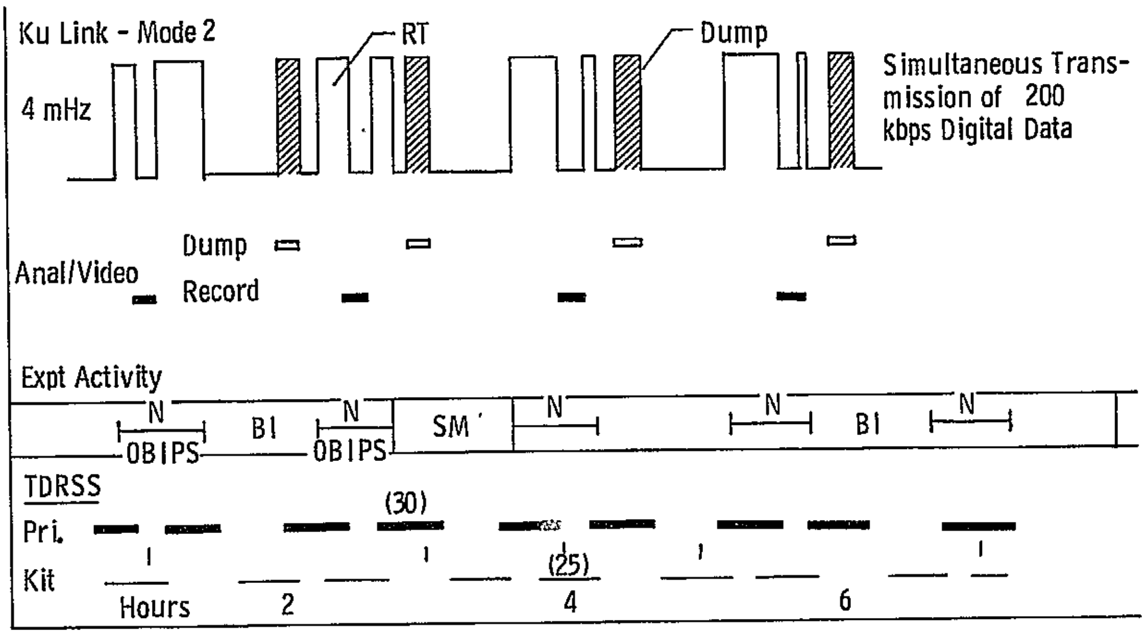


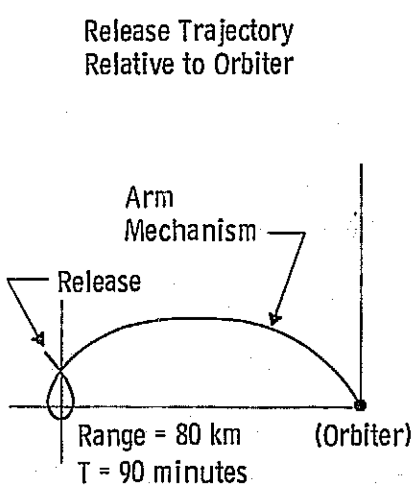
Figure 4.7-8 Data Recovery Timeline C

These analyses indicate the compatibility of the proposed single Ku antenna concept, and also indicate the need for greater DOMSAT bandwidths to accommodate the 15.2 Mbps digital peak load and the 4 MHz peak video signal.

The Orbiter Ku system will also be used in the radar mode to track instrument packages deployed from the payload bay. It is intended that tracking will only be required for a period of 5-10 minutes following deployment so that the processed radar data can be used to update the predicted package trajectory. On this basis, there will be only minimum interference with Ku band communications, and only at times when no other significant experiment activity is scheduled. The link analyses for these applications are presented in Section 5.8.2.

Gas Release Communications - A series of six gas releases are deployed from the payload bay on a one per day basis, and follow the trajectory shown in Figure 4.7-9. Each release requires the transmission of a small number of commands to arm a mechanism in mid-flight

and release the gas at a range of approximately 80 Km. The Orbiter detached payload link was first evaluated for satisfying this command requirement. As indicated in Figure 4.7-9, the Orbiter EIRP would require a high quality command receiver with a noise figure below 4 dB to achieve the minimum desired 12 dB carrier-to-noise (C/N) ratio. Since this approach was not considered to be cost effective, another concept was analyzed and selected. This concept utilizes a dedicated AMPS command transmitter with a 10-12 watt output at S-band. The detailed link analyses described in Section 5.8.2 indicate that available command receivers with a 12 db noise figure will accommodate this requirement and be very cost effective for non-retrievable packages. Additional cost savings may be achieved if tone commands can be utilized effectively, thus reducing the cost of the command decoders. In addition to the FM transmitter, the AMPS RF terminal would use a conical spiral antenna to be shared with other communication links to deployed packages. The gas release RF system consists of a stub antenna and the command receiver.



Option A - Orbiter Payload Link

EIRP (+31 dBm) over 80-km Range Requires High-Quality Command Receiver -

R_S = -155 dBw, NF < 4 dB

Digital Command Decoder

Option B - Dedicated AMPS Command Transmitter

P_O = 10 W Provides RF Signal Compatible with

Typical Receivers

Use of Tone Commands

Cost Effective Concept for Nonretrievable Releases (6 Units)

Same Comments Apply to Chemical Release

Figure 4.7-9 Gas Release Communications

ESP Communications - The ESP is first deployed by the RMS to do EMI mapping of the payload bay, and is then ejected and follows a trajectory trailing the Orbiter by no more than 4 Km. In addition to a digital command link, recovery of 16 Kbps of data is required while attached to the RMS and throughout the ESP trajectory. At first, it was thought that this requirement was compatible with the Orbiter detached payload link. However, closer evaluation of the Orbiter antenna

coverage revealed a possible problem, which is illustrated in Figure 4.7-10. This figure shows the Orbiter antenna located in the cockpit area and the extent of its 100 degree, -3 dB beamwidth. Recent information indicates that this antenna pattern will be as closely restricted to this ±50 degree coverage as possible in order to avoid interference with other Orbiter S-band antennas also mounted in the cockpit area. As the figure indicates, this limited coverage restricts the RMS movement if communications to an RMS package is to occur. RMS movement to an area beyond the first pallet will not be possible, thus limiting payload bay EMI mapping by the ESP.

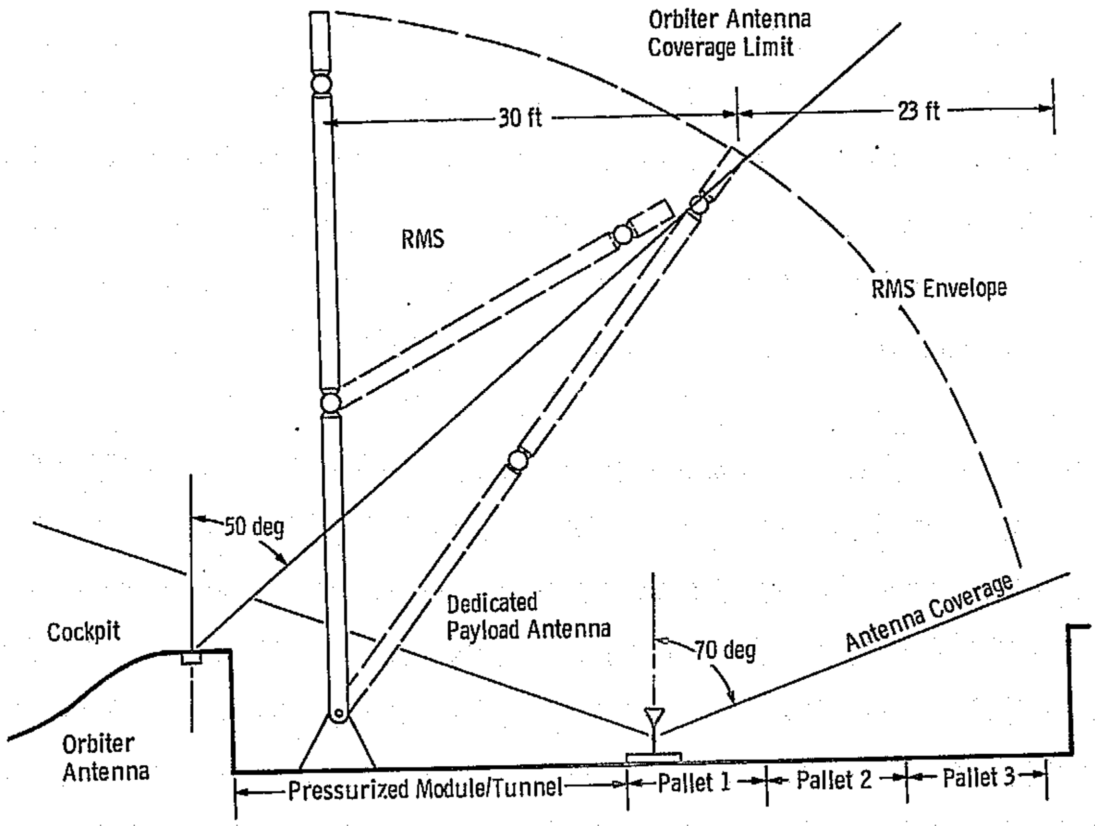


Figure 4.7-10 Orbiter Bay Antenna Coverage

Our selected concept, therefore, shares the conical spiral antenna used by the gas release described earlier as well as the command transmitter modulated with digital commands. Additionally, an FM data receiver and diplexer will complete the AMPS RF terminal.

The ESP RF hardware will consist of a 1 watt FM transmitter (S-band), a command receiver similar to those used on the gas releases,

a diplexer-power splitter combination, and two conical spiral antennas mounted on opposite faces of the ESP to provide nearly omni-directional coverage. A diagram of this system will be shown in a later figure; and the link analyses are presented in Section 5.8.2.

RMS Package Communications - A trade study was performed to compare methods of data transfer between the pallets and RMS deployed packages. Some non-standard techniques for cable deployment and control proved to be more costly and involve greater development risk than the conventional RF data interface. Only one instrument package used for Beam Diagnostics is deployed during Flight 1, in addition to the previously mentioned ESP. The RF system required on the Beam Diagnostics package is shown in Figure 4.7-11, along with the Flight 2 RMS packages. It requires a two-way data transfer, with the data link requirements including a video signal and a wideband combination of analog and digital data modulating two RF carriers. Since the propagation distance involved is never more than 25 meters, a 100 milliwatt transmitter output is sufficient in conjunction with a simple, low gain stub antenna. The command receiver will be the same model as described for the releases and ESP; and this will probably require temporary attenuation of at least 20 dB of the AMPS command transmitter output. The stub antenna will be mounted so that its typical hole in the pattern will not point toward the payload bay. As can be seen from Figure 4.7-11, this link will also share the AMPS RF terminal, which is now revised to include two wideband receivers.

Flight 1 Communications Configuration - The overall flight communications configuration is diagrammed in Figure 4.7-12, and it satisfies all communication requirements presented previously in Table 4.7-1. The three separate applications shown are timeline and functionally compatible. Only one instrument package of those shown is on at a time. The single command carrier transmits commands to each package; and the ESP return carrier frequency will be the same as one of the two used by the RMS package.

Particular note should be taken of the cross-hatched boxes which are indicative of RF components that are reusable on Flight 2. This approach is very compatible with the Labcraft philosophy in that these RF components can become a bank of telemetry and command system hardware available for repeated use on Labcraft payloads.

For the most part, the RF transmitters and receivers are available commercial hardware previously used for rocket flight testing. Their specifications are quite compatible with the payload environments described in the Orbiter Payload Accommodations Handbook.

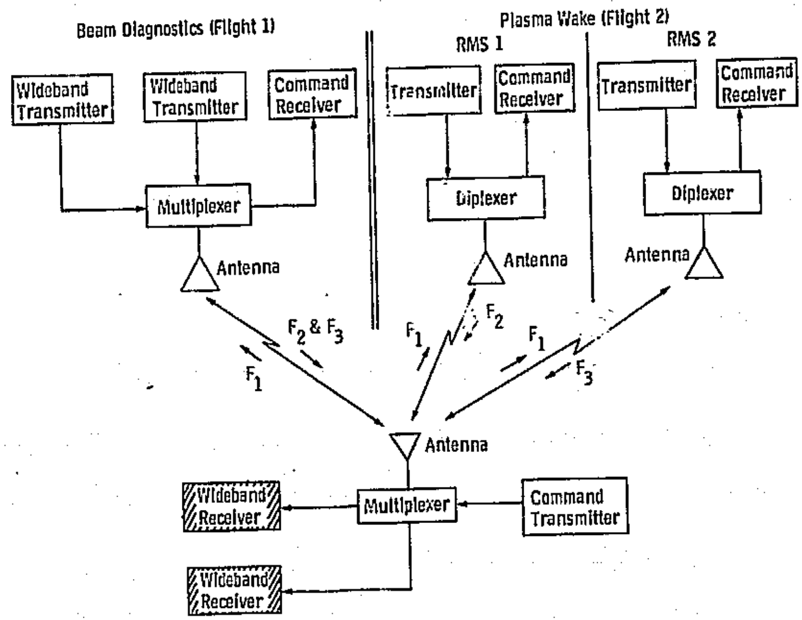


Figure 4.7-11 RMS Communications

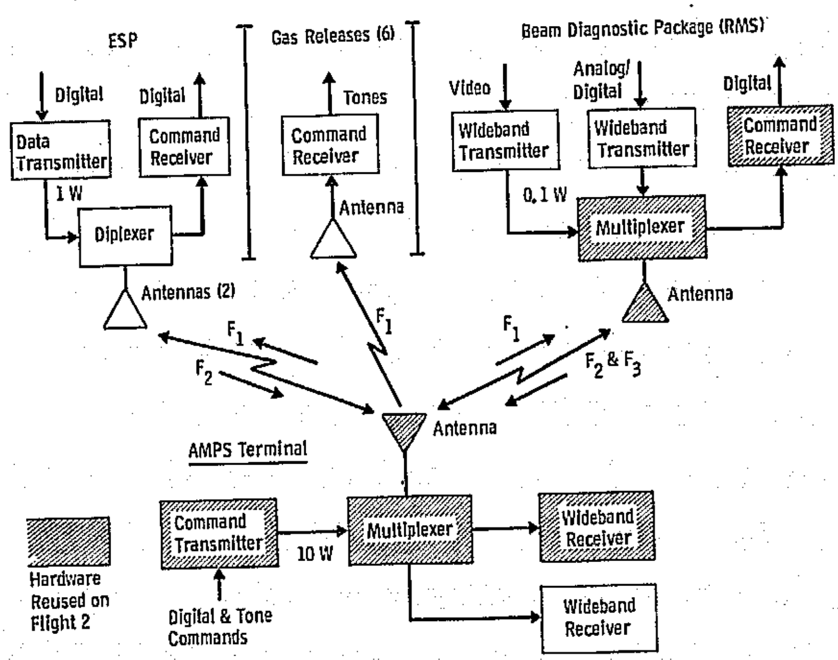


Figure 4.7-12 Flight 1 Communications Configuration

4.7.3 Flight 2 Communications Concepts

This section will describe various options that were evaluated for satisfying the Flight 2 communication requirements listed in Table 4.7-1. Where there are similarities to Flight 1 requirements, only the differences will be discussed in detail. As indicated previously, the tracking requirement associated with deployed packages is satisfied by the Orbiter Ku radar. The Orbiter-to-ground communications concept presented in Section 4.7.2, which utilizes only one Ku antenna, is also feasible for Flight 2.

RF Receiver Package Communications - The RF Receiver Package is ejected from the payload bay in a spin stabilized mode, and follows the trajectory illustrated in Figure 4.7-13, during which time propagation and sounding experiments are being conducted. Control of the experiments requires a digital command link from the payload bay and the retrieval of combined digital (4 Kbps) and analog (30 KHz) data. Link analyses associated with this requirement are presented in Section 5.8.2. For the same reasons as the gas release command link, the Orbiter cannot satisfy this requirement. However, the 10 watt output of the AMPS RF terminal described for Flight 1 provides a satisfactory +15 dB C/N. Because of the need to recover analog and digital data, which the Orbiter cannot handle, the AMPS RF terminal must establish a two-way interface with the Receiver Package.

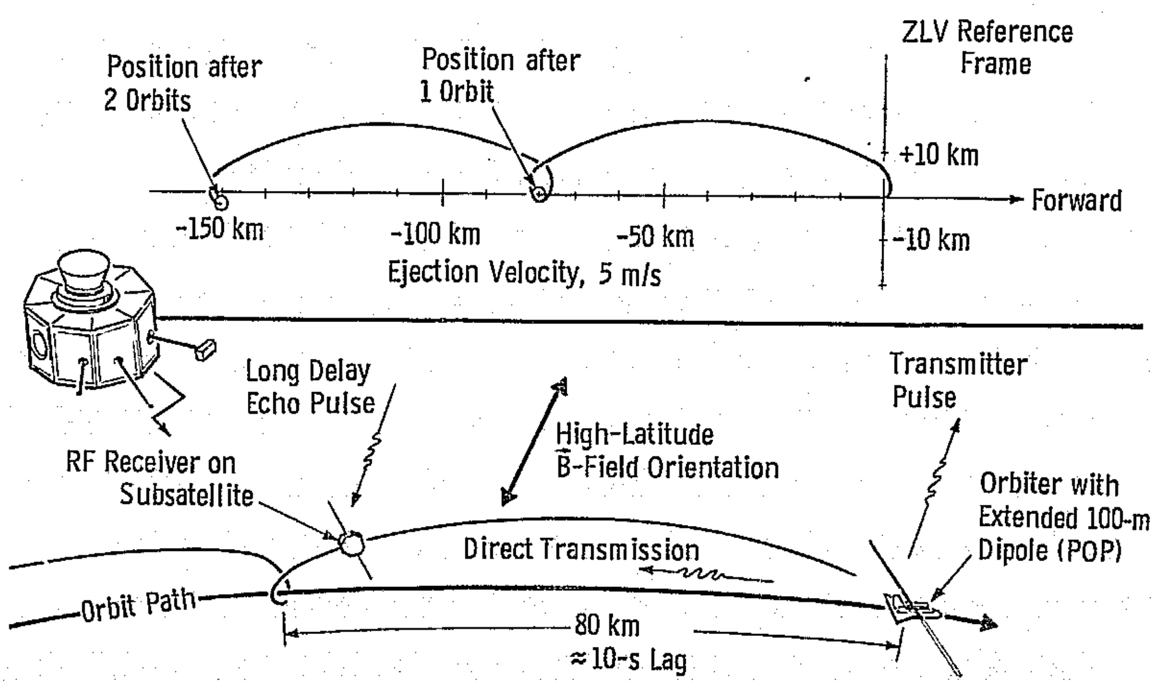


Figure 4.7-13 RF Receiver Trajectory

RF components required onboard the RF receiver include a 10 watt, FM transmitter whose signal is split between two conical spiral antennas providing nearly omni-directional coverage for the rotating package. A multiplexer-power splitter and FM command receiver complete the RF system. Only a single RF carrier is used in conjunction with a subcarrier oscillator to handle both data signals.

RMS Communications - One experiment scheduled during Flight 2, Plasma Flow, requires RMS deployment of instrument packages supported by RF communication links. In this case, packages are deployed on both RMSs, with each requiring a two-way command and data link to the AMPS RF terminal, as pictured previously in Figure 4.7-11. As the figure confirms, these requirements are compatible with the AMPS RF terminal in that two RF downlink carriers are required for moderate data rates, in addition to a single command carrier with a dual address capability.

Flight 2 Communications Configuration - The Flight 2 communications configuration is shown in Figure 4.7-14, and it satisfies all requirements identified in Table 4.7-1. As was the case for Flight 1, a review of the Flight 2 timeline indicates no conflict in that no two of the applications are active simultaneously.

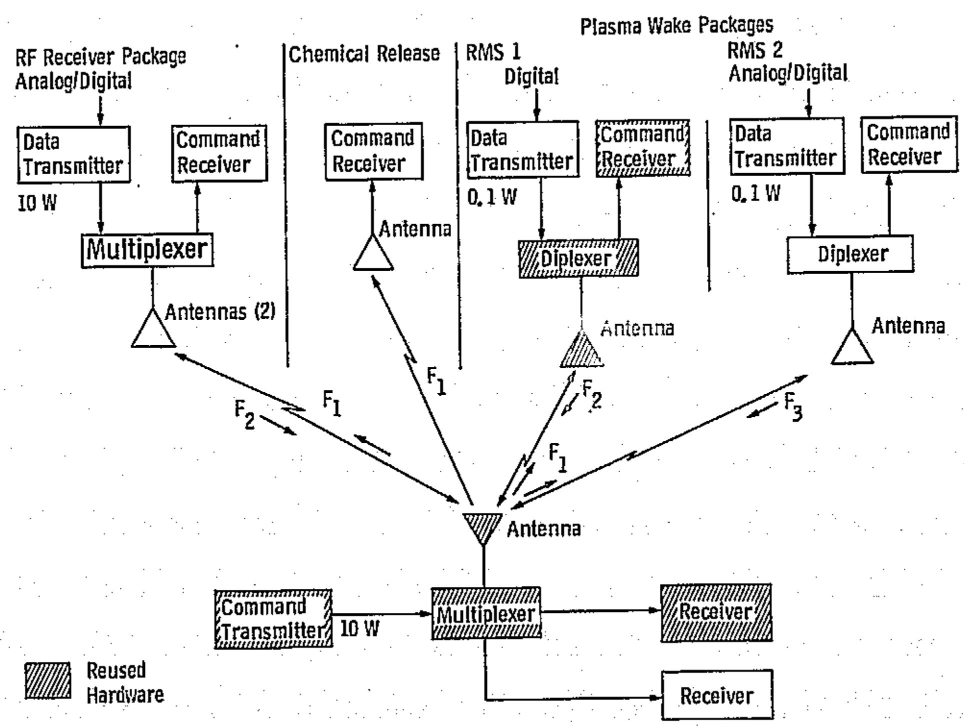


Figure 4.7-14 Flight 2 Communications Configuration

The figure also indicates by cross-hatched boxes those units re-used from Flight 1, and along with nearly all the Flight 2 hardware, is now available for use on later flights.

4.8 Deployed Instrument Support Subsystems

Flights 1 and 2 of AMPS include six experiments which require that a package containing instruments and other subsystems be deployed out of the Orbiter cargo bay. Some packages are held with the Remote Manipulator System (RMS) while others are released from the RMS to free fly. Several packages are ejected with a specific delta velocity and direction relative to the Orbiter. The following paragraphs describe the six deployed experiments and the deployed packages. This section concludes with a summary of a maneuverable subsatellite analysis.

4.8.1 Environmental Sensing Package (ESP)

The ESP is a modular grouping of instruments and subsystem support equipment that, in conjunction with Orbiter/Spacelab, performs experiments in EMI measurement and Orbiter wake mapping. The objectives are to obtain quantitative empirical values of both the near and far field EMI environment of the AMPS payload, and to map the close-in and far-field wake characteristics of the Orbiter. Data from the latter experiments are used as baseline information for planning plasma wake experiments on future flights.

The ESP operates in three various modes after controlled deployment with the RMS or ejection from the payload bay. During operational Mode 1, close-in EMI mapping is performed with the ESP supported and manipulated by the RMS. In Mode 2 the module is extended by the RMS for Orbiter wake mapping while the Orbiter performs a roll maneuver at 4 rpm. Mode 3 involves remote operation of the module after ejection from the end of the RMS for continued Orbiter wake and EMI far-field measures. Between modes the ESP is re-stowed in the pallet mount. Section 3.4 discusses the detailed operation of these experiments.

The basic structural module is eight-sided with instruments mounted on the periphery (Figure 4.8-1). Subsystem support equipment is in the center section, with the exception of a battery that is mounted in an outside compartment because of size and thermal considerations. Mechanisms include the 4 rpm spin system, the separation/eject system, the capture/release mechanism, and deployment devices for antennas and sensors. The mechanism and structural details are discussed in Section 4.1

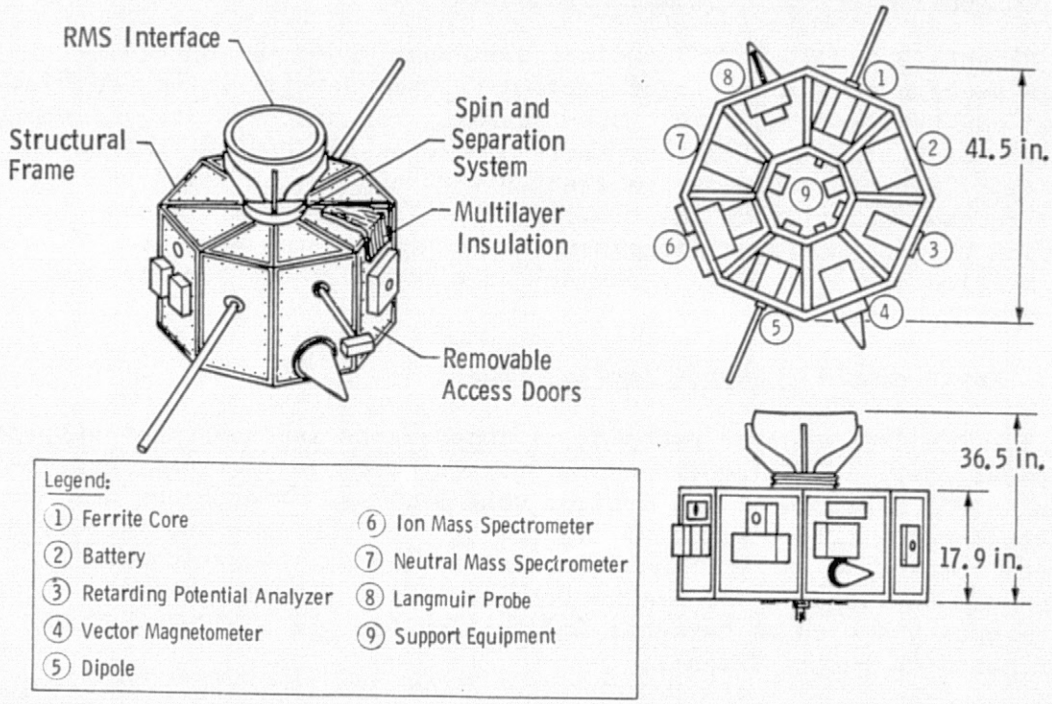


Figure 4.8-1 ESP Configuration

Table 4.8-1 lists the instruments mounted in the ESP, the operational support equipment and the weight breakdown. Communication and data management are by RF link with self-contained power supply. Thermal control is cold bias with heaters and multilayer insulation with selected radiation areas. Electrical power is provided by a 28V, 105 ampere-hour battery which supplies 163 watts. The telemetry subsystem requires commands to the package and data return to the Orbiter. The maximum data rate is 16K bits per second.

4.8.2 Beam Diagnostics Package

The Beam Diagnostics Package consists of an arrangement of instruments and support equipment that are grouped together to perform the Electron Beam Studies experiment. The experiment objectives is to study electron beam structure and stability along with studies of beam interaction with the ambient and induced atmosphere. This investigation is performed in two modes, both of which use the electron accelerator. In the first mode, a plume of nitrogen gas is released above the accelerator. As the plume disperses, the beam is fired through it. Interactions of the energetic electrons

Table 4.8-1 Environmental Sensing Package Weight Breakdown

Item	Weight (LB)
Instruments	
Vector Magnetometer	11.2
Retarding Potential Analyzer	6.6
Neutral Mass Spectrometer	22.0
Dipole Antenna	11.0
Ferrite Core Antenna	8.8
Langmuir Probe	11.0
Ion Mass Spectrometer	11.0
Subsystems Support Equipment	
Communications	
Transmitter	1.1
Receiver	1.3
Diplexer	2.2
Conical Antennas	4.4
Data	
PCM Programmer	4.4
Decoder	2.2
Power	
Battery and Cables	55.1
Command Initiator Module	1.1
Thermal	
Strip Heaters	2.2
Multi-layer Insulation	8.6
Structure and Mechanisms	143.1
TOTAL	307.3 (139.4 Kg)

with the nitrogen molecules give a visible indication of the beam structure and characteristics. In the second mode, vehicle potential measurements are made and beam direction with respect to the magnetic field are monitored by instruments inserted into the beam. Operational details are shown in Section 3.4. Three instruments are used to monitor and observe the beam. OBIPS (II-3) is used to visually observe the beam structure and the orbiter surfaces for discharge effects. Level II Beam Diagnostics (III-4) and the Vector Magnetometer (III-2) map the beam cross-section at various elevations above the accelerator.

To accomplish the science functions, the Beam Diagnostics Package must be deployed above the payload bay by the RMS. For visual observations, the package should be deployed as far away from the beam as possible. For Mode 2 operations, a sweep across the beam is required.

Figure 4.8-2 shows the overall module configuration that was derived during the design studies. The module, as shown, contains power, data, communications and thermal subsystems in addition to the instruments. A square boom type frame is utilized to house the instruments and support equipment and provides added length to allow deployment over the accelerator on the aft pallet. In addition to the basic framework, the structures and mechanisms subsystem features: an RMS and effector interface, a capture/release device interface, a launch/landing lock system for the vector magnetometer sensor, and the equipment mounting brackets.

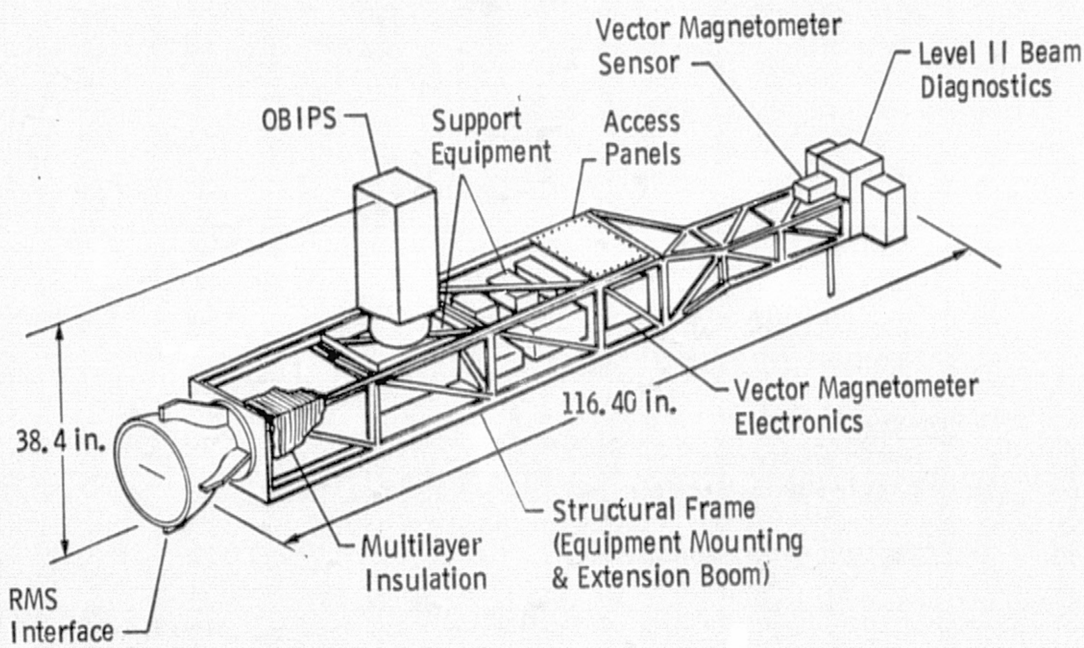


Figure 4.8-2 Beam Diagnostics Package

The beam diagnostics package uses an RF link for transmission and reception of data, command and control signals. An onboard power supply consisting of a 28V, 160 ampere-hour battery supplying 163 watts, power distribution unit and an interconnecting cable set are provided. Video, digital and analog data plus commands are required

to support the deployed module. These requirements are the most demanding of any of the deployed modules. The video signal (4 MHz) requires a dedicated transmitter because of the data bandwidth. A separate transmitter handles the analog and digital data after processing by a PCM Programmer and a Subcarrier Oscillator assembly. A Command Decoder provides on-board commands to the instruments and support equipment. The communication subsystem is made up of a Command Receiver, RF Multiplexer, and a stub antenna in addition to the two transmitters. The return data is sent to the pallet-mounted RF terminal at a rate of 90 K bits per second using five channels (100 KHz per channel). Thermal control is provided using a cold-bias with heater design. Strip heaters powered by the battery or the Spacelab bus are mounted on the main frame members to provide additional heat during nonoperating periods. The use of multi-layer insulation along with silver coated teflon radiation areas, allows an environmental balance to be maintained under all conditions. A listing of the component make-up and weight breakdown is presented in Table 4.8-2.

Table 4.8-2 Beam Diagnostic Package Weight Breakdown

Item	Weight (LB)
INSTRUMENTS	
OBIPS	83.8
Vector Magnetometer	9.0
Level II Beam Diagnostics	50.7
SUBSYSTEMS EQUIPMENT	
Communications	
Wide Band Transmitter (2)	2.2
Command Receiver	2.2
RF Multiplexer	2.2
Antenna (Stub)	2.2
Data Management	
Command Decoder	3.3
PCM Programmer	4.4
Subcarrier Oscillator	11.0
Thermal	
Heaters	2.2
Multi-Layer Insulation	8.8
Power	
Power Supply	63.9
Cable Set	3.3
Structures and Mechanisms	66.6
TOTAL	315.8 (143.3 kg)

4.8.3 RF Receiver Package

This deployed package contains instrumentation required to perform electromagnetic wave propagation and interaction experiments. The package is deployed from the Orbiter in a spin stabilized mode (4 rpm.). It follows a trajectory with respect to the Orbiter as illustrated in Section 3.4 to a distance of about 80 Km or more over which RF transmission and long delay echo transmissions are made. The package configuration as shown in Figure 4.8-3 indicates the instrument and support equipment location. The RF Receiver instrument complement consists of the RF Plasma Wave Receiver and Antenna, and the Fluxgate Vector Magnetometer which operate in conjunction with the RF Plasma Wave Transmitter, Receiver, and 100 meter Dipole Antenna located in the payload bay. Operational details of experiment performance are discussed in Section 3.4.

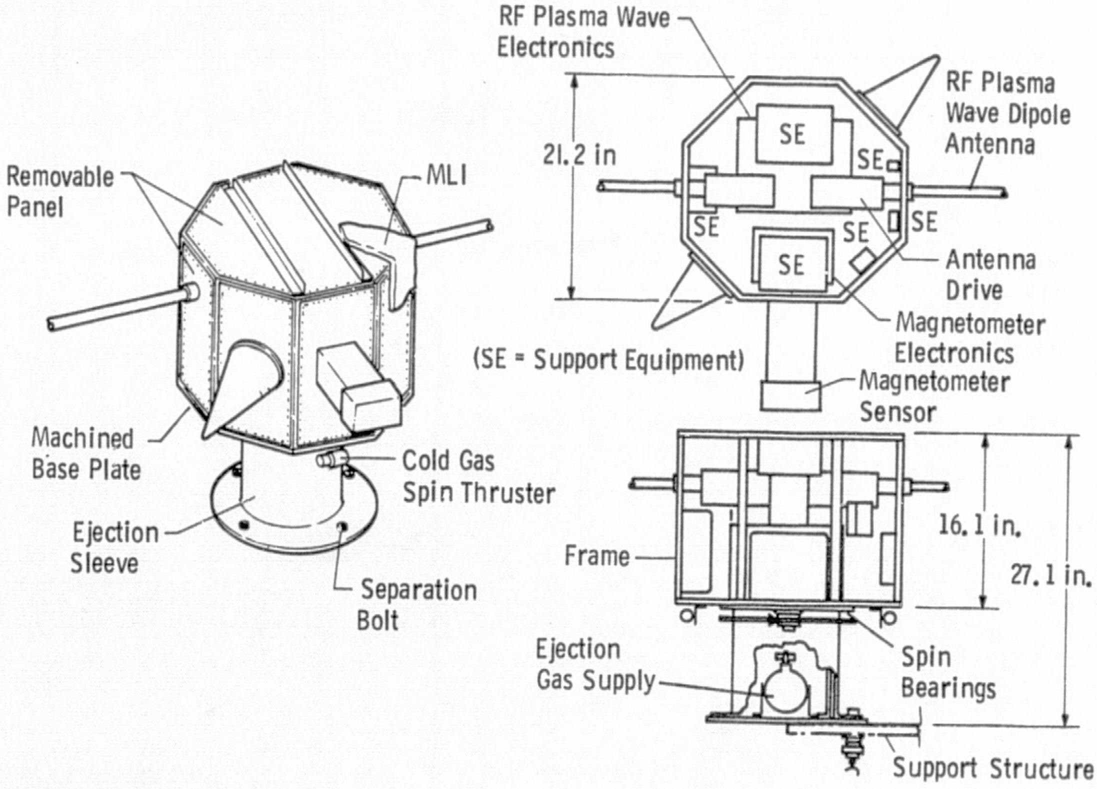


Figure 4.8-3 Receiver Package

Implementation of the experiment uses two modes. The first mode uses a single-ended sounder technique; the second mode uses a receiver located on a subsatellite. Mode 1 requires a transmitter and receiver coupled with a 100-meter tip-to-tip antenna. Orientation of the antenna with respect to the geomagnetic field is monitored by a fluxgate vector magnetometer deployed on the RMS. The instrument is operated in a sounder mode over a range of conditions including various orientations with respect to the geomagnetic field vector, high and middle latitude and day/night variations.

Mode 2 requires the same transmitter and dipole antenna on the Orbiter and also requires a receiver and a 10-meter tip-to-tip dipole antenna on the subsatellite. Attitude of the antenna with respect to the geomagnetic field vector is monitored by a fluxgate vector magnetometer. During this mode of operation, propagation-type measurements are taken using the Orbiter-mounted instruments as the source with the receiver on the subsatellite. After the subsatellite is deployed from the Orbiter, the total observation time available is on the order of one to two orbit periods. This operating period will also be shared with the long-delay echo experiment as noted in Section 3.4.

Support systems required onboard the RF Receiver Package include a battery power supply data acquisition and communications equipment, and a small amount of thermal control hardware.

The RF Receiver Package receives thermal control power from the Spacelab power subsystem while it is mounted on the Spacelab pallet. The power required for the operation of the entire package during free-flight is provided by a silver-zinc primary battery. The battery is sized to provide power for all package subsystems for two hours after release of the package from the RMS. The total energy required for two hours of operation is 26 ampere-hours at an average discharge rate of 4 amperes. The battery selected is a 65 ampere-hours design which gives a 100 percent energy margin.

The RF Receiver instruments generate about 4 Kbps of digital data and a 30 KHz analog signal which require transmission to the payload bay; and they also require approximately 45 unique commands for control of the package. The onboard digital data is acquired using a multiplexer and PCM encoder which, in turn, provides a PCM signal to modulate one of two subcarrier oscillators. The other oscillator is modulated by the 30 KHz analog signal; and the two oscillator outputs are mixed before modulating an S-band FM transmitter. The RF system consists of a 10 watt transmitter whose output signal is split between two conical spiral antennas on opposite faces of the RF Receiver Package. A diplexer and command receiver-decoder accepts the S-Band transmission from the AMPS transmitter in the payload bay. Following its ejection, the RF Receiver Package is skin tracked by the Orbiter K_u radar, but this requires no onboard hardware. The

heat dissipation levels of the onboard hardware is such that only a basic Thermal Control Subsystem (TCS) is required. The TCS components include multilayer insulation and heaters; and operate in cold-biased configuration. A listing of the component make-up and weight breakdown is presented in Table 4.8-3.

Table 4.8-3 RF Receiver Package Weight Breakdown

Item	Weight (LB)
Instruments	
Vector Magnetometer	9.0
RF Plasma Wave	15.4
Subsystems Equipment	
Communications	
Transmitter	1.1
Command Receiver	1.3
Diplexer	2.2
Antenna, Conical (2)	4.4
Data	
Command Decoder	2.2
PCM Programmer	4.4
Subcarrier Oscillator	2.0
Power	
Power Supply	22.0
Cable Set	2.2
Thermal	
Strip Heaters	2.2
Multi-layer Insulation	3.7
Structures & Mechanisms	58.6
TOTAL	130.7 (59.3 kg)

4.8.4 Gas Release Module

The objective of the experiment is to study the source mechanism, propagation characteristics, and other properties of naturally occurring gravity waves in the earth's atmosphere by generating an artificial gravity wave. The plan calls for the release of a significant amount of neutral gas (70 kg of nitrogen) to generate waves in the upper atmosphere, and to observe the evolution of the resulting wave field.

The gas is released by a module ejected from the payload bay and deployed at a safe separation distance. The release takes less than a second and provides a virtual point source for the rapidly expanding gas cloud as it interacts with the ambient atmosphere and comes to a halt. The momentum/energy pulse generates a propagative wave field observed by the backscatter radar of the Arecibo Inospheric Obervatory and a three-station ground network of RF sounder. The release conditions which require a quiet background for optical viewing are shown in Section 3.4 as well as the target release zones and the Orbiter viewing constraints.

The experiment, performed six times on the first flight, requires six gas release modules. The distance of the gas release from the ground radar will be varied as follows: two at 0 km, two at 10 km, and the final two at 200 km. The size of the release and the altitude are to be held constant for each experiment period. The initial phase of the gas cloud expansion is optically observed both from the ground station and from the OBIPS located on the rear MPM. The release is seeded with an optically active material to make the release visible to both stations (ground and payload bay), which are equipped with low-light-level (L^3) TV cameras.

The gas release module consists essentially of a tank with gas release system, equipment module, ejection system, thermal control system, and support structure as shown in Figure 4.8-4.

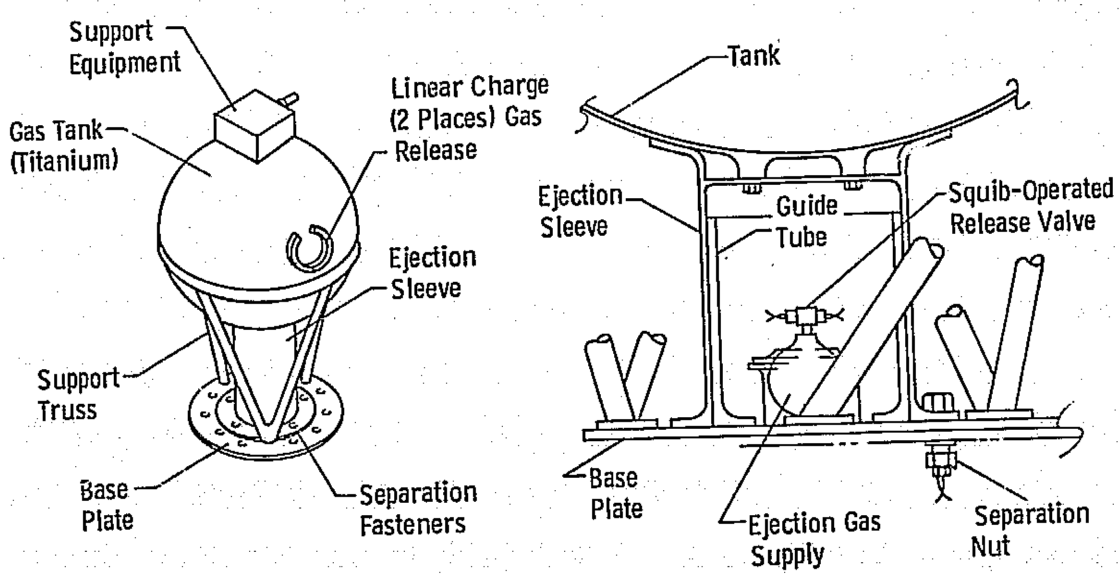


Figure 4.8-4 Gas Release Module

Each of the six tanks is approximately 33 inches in diameter and consists of a 3300 psi (6600 psi burst) titanium sphere containing nitrogen. Two linear charges are located diametrically opposite each other on each tank. When detonated, these charges will cut two holes in the tank as indicated in the figure (side view). The hinged flaps prevent any debris from interfering with the system operation.

The thermal control system consists of multi-layer insulation with internal strip heaters around the tank and equipment module. Support structure is wrapped with silver coated teflon. Power is supplied to the strip heaters from the Orbiter when the tanks are in the payload bay. Prior to ejection, either power is removed from the strip heaters and a cable separator squib is initiated, or a fly-away disconnect plug may be used; the latter implementation is recommended.

The support structure consists of trusses, a base plate and a guide support structure which remains in the payload bay at tank ejection. The tubular truss (pipe/round tube) forms a circular seat onto which the tank is preloaded by bolting down the outer sleeve to a base plate.

Located below a large gas tank is the ejection system which consists of a second smaller gas tank/chamber, an ejection guide tube/sleeve, and three separation nuts. After the separation nuts have been fired, the cold gas is released from the latter tank by a squib operated valve (solenoid device) into a cylindrical chamber consisting of two concentric sleeves (guides). The gas pressure reacts on the large tank through the outer sleeve's top closure and outer sleeve. The tank starts to eject by sliding up and off the inner sleeve.

An equipment module located on top of each gas tank contains electronic modules, antenna and a 28 V, 1.7 ampere-hours battery for the communication and gas release systems. The communication system consists of the command receiver, command decoder and a stub (omni) antenna. The gas release system consists of an initiator command module, switching relay, and two linear charges previously described. A battery supplies the power for both systems. The initiator command module consists essentially of a voltage converter, a storage capacitor, triggering (fire) circuit, and a monitor circuit. To initiate a gas release, a discrete trigger command is sent to the switching relay which has previously received a power-on (arm) command to activate the initiator command module. The trigger command enables the firing circuit to detonate both linear charges. A listing of the component makeup and weight breakdown is presented in Table 4.8-4.

Table 4.8-4 Gas Release Module Weight Breakdown

Item	Weight (LB)
Instruments	
Gas Release (tank and gas)	348.2
Subsystems Equipment	
Communications	
Command Receiver	2.2
Antenna, Stub	2.2
Data	
Command Decoder	3.3
Power	
Power Supply	6.6
Cable Set	3.0
Initiator Command Module	1.1
Thermal	
Strip Heaters	2.2
Multi-layer Insulation	3.5
Structures and Mechanisms	39.4
TOTAL	411.7 (186.7 kg)

4.8.5 Chemical Release

The objective of this experiment is to modify the natural ionospheric and magnetospheric currents by changing the ionospheric conductivity on a relatively large scale.

A neutral barium cloud is green and an ionized barium cloud is red. Observation of the relative motion of the two clouds is used to determine the structure of the ionospheric and magnetospheric currents. The resulting disturbance of the barium clouds may trigger an auroral display. It is therefore desirable to release the barium in a region of maximum auroral frequency. The barium cloud required for this experiment is cylindrical in form, at an altitude of 150 to 200 km, 15 to 20 km in diameter and over 100 km long. Generation of a cloud this size requires 100 kg of barium thermite reaction at a uniform rate of 10 kg/km. The orbiter covers approximately 100 km in 12 seconds.

The chemical release experiment is performed in a three-orbit sequence. The module containing the barium thermite is removed from the pallet by the Orbiter RMS and deployed with zero velocity relative to the Orbiter, over the Hudson Bay region. The Orbiter is then maneuvered away from the module, requiring one orbit to move the necessary 80 km distance. The barium vapor is generated at a uniform rate over the 100 km release zone. The barium release should occur with clear skies over the ground station and timed as follows:

(1) Near sunset terminator to permit observation through the night.

(2) When zenith at visual ground stations is greater than 96 deg from the sun line to permit good viewing.

(3) When nadir from release cloud is less than 99 deg from the sun to permit adequate photoionization to occur.

Ground observations at Fort Churchill throughout the night include: visual observations of auroral phenomena; use of radio frequency sounders; fluctuations in the electric and magnetic fields; and sounding rocket probes. The next orbit pass 200 to 400 kilometers south of the release zone affords excellent viewing. The on-board low-light-level TV and the two Ebert Fastie spectrometers are used to observe the auroral processes.

Typically, the barium thermite is loaded in high pressure, insulated canisters. An available canister built by Thiokol, has been baselined for use on AMPS and is illustrated in Figure 4.8-5. The canister includes a pyrotechnic igniter and a nozzle assembly. Nozzles can vary in number and configuration depending upon particular application. Each Thiokol canister contains 16 kg of barium thermite. Sixty-four canisters are needed to provide 1024 kg of thermite. The barium thermite reaction for each canister is completed in approximately 1.5 seconds. Proper sequencing of these firings results in the uniform release of the barium in the required 12 second interval.

The Thiokol canisters have been packaged in a rectangular welded frame structure as shown in Figure 4.8-6. The canisters are mounted symmetrically and are fired in a manner so as to minimize propulsive effects. A discrete signal command receiver, battery, decoder and sequencer system provides for control of the release process. There is no return data from the canisters. Thermal control of the module is provided by external multi-layer insulation and selected radiation areas of silver coated teflon. Strip heaters are provided for cold case operation. The electrical power required for the deployed package is provided by self-contained 1.7 ampere-hour batteries. These provide 65 watts of 28 VDC. A listing of the component makeup and weight breakdown is presented in Table 4.8-5.

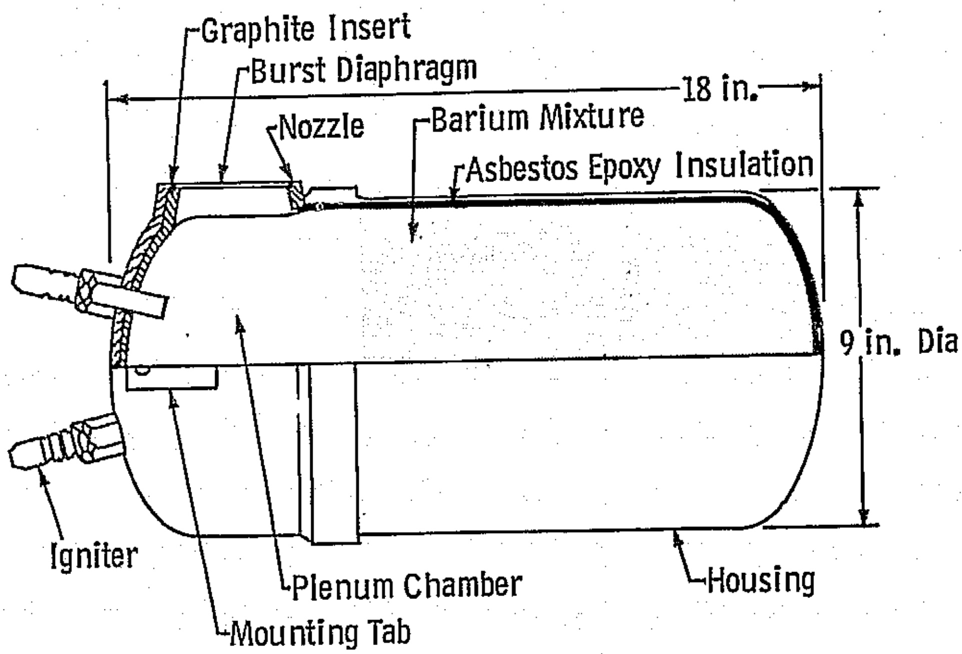


Figure 4.8-5 Thiokol Barium Thermite Canister

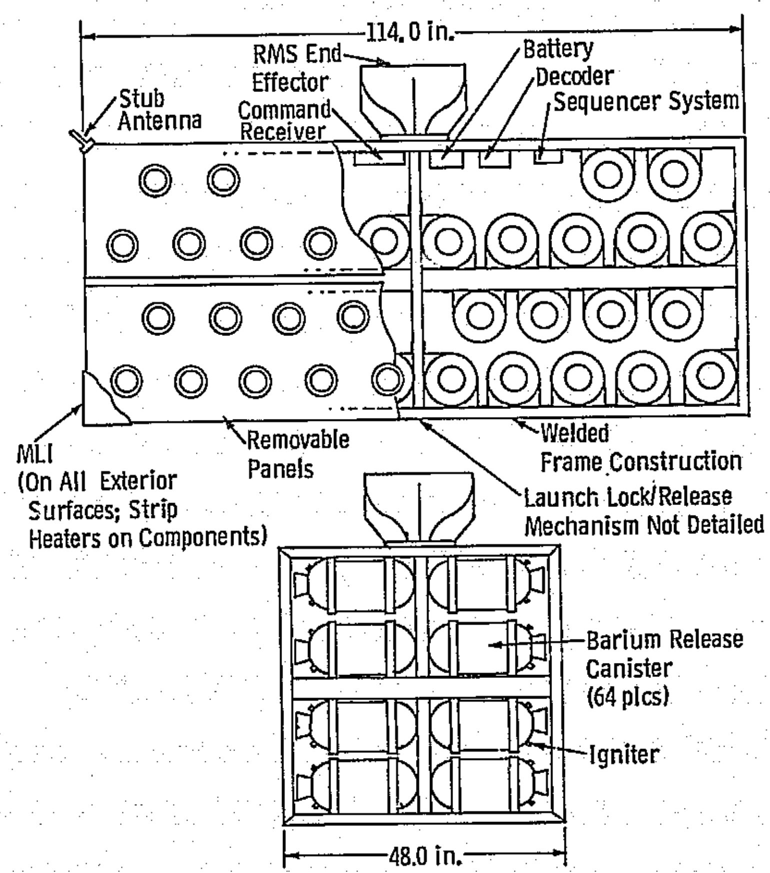


Figure 4.8-6 Chemical Release Module Configuration

Table 4.8-5 Chemical Release Weight Breakdown

Item	Weight (LB)
Instruments	
Chemical Release	3245.2
Subsystems Equipment	
Communications	
Command Receiver	1.3
Antenna, Stub	2.2
Data	
Command Decoder	3.3
Sequencer	22.0
Power	
Power Supply	6.6
Cable Set	5.5
Pyro-Initiator Command Module	4.4
Thermal	
Strip Heaters	2.2
Multi-layer Insulation	38.6
Structure and Mechanisms	344.2
TOTAL	3675.2 (1667.2 kg)

4.8.6 Plasma Diagnostic Package

Using this package the objectives of the experiment are to study the characteristics of the plasma wake generated by an axially symmetrical test body whose surface must be conductive and whose potential relative to the ambient space environment is maintained at a known level. The wake is generated by a 4-meter metallized balloon mounted on the end of a 5-meter extension boom which in turn is deployed on the end of one RMS, as shown in Section 3.4. The Orbiter is oriented so the inflated sphere is the most forward surface in the velocity direction. The orientation is maintained as the Orbiter moves through various plasma conditions: middle and high latitudes; the South Atlantic anomaly region; and day/night variations. The potential of the sphere with respect to the Orbiter is maintained by a bias power supply included in the instruments.

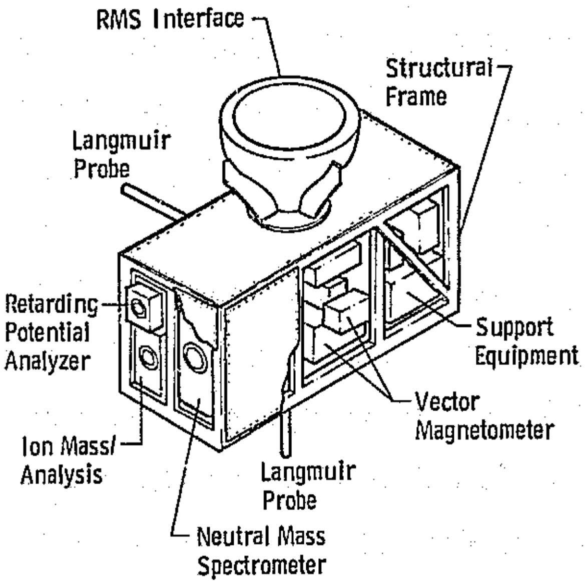
During these various orbital movements, the wake characteristics are monitored by a package of diagnostic instruments deployed by the second RMS so the wake region can be spatially scanned. The instruments included in the diagnostic package are indicated in Table 4.8-6.

Table 4.8-6 Instruments For The Plasma Flow Experiment

Instrument	IFRD No.	Measures
Vector Magnetometer	III-2	Magnetic field
Ion Mass & Distribution Analyzer	III-10	Thermal ion density, temperature drift velocity
Planar RPA	III-18	Distribution function of thermal electron population
Langmuir Probe	III-22	Ambient electron temperature, density and space potential.
Neutral Mass Spectrometer	III-23	Neutral gas composition

These instruments were packaged along with support equipment as shown in Figure 4.8-7. The primary requirements governing the design are as follows:

- The Langmuir Probe must be parallel and perpendicular to the velocity vecotr.
- The RPA, Ion Mass Analyzer and the neutral mass spectrometer must face the velocity direction.
- Provisions for deployment of the vector magnetometer must be provided.
- Capability must be provided to deploy and restow the diagnostic package.
- Equipment layout must consider thermal constraints.
- Provisions must be made for data/communication and power.



Science Drivers/Requirements

RMS Deployed and Restowed
Velocity Vector Orientation

Packaging Concept/Physical Characteristics

Structural Frame with Equipment Mounting Brackets
Removable Access Panels
Multilayer Insulation (Cold Bias with Heaters)
Size: 15.0 x 18.0 x 30.4 in.
Weight: 58 kg

Mechanisms

Capture/Release Device
Sensor Deployment Mechanism

Figure 4.8-7 Plasma Wake Diagnostic Package

Thermal design includes a multi-layer insulation (cold bias with heaters) to maintain a temperature environment of 0 to 120^{o}F during operation. The external surface also includes silver teflon radiation areas that reduce orbital temperature variations and also provides a low temperature thermal sink.

A power supply and data/communication support equipment are provided. The electrical system consists of a 28VDC, 65 ampere-hour storage battery providing 143 watts, a power distribution unit and an interconnecting cable set. The storage battery is activated prior to installation on the Diagnostics Package and remains in a ready state until power is required. Thermal heating is supplied from the Spacelab power bus prior to package deployment. The data command and communication subsystem makes maximum use of equipment flown on Flight 1. The PCM encoder is reformatted for a nominal data rate of 100 Kbps. Three channels of analog data are FM multiplexed and the combined digital and FM data are in turn multiplexed via the subcarrier oscillator. A low powered transmitter of no more than 0.1 watts is required for data transmission since the RF distance is small. The command system uses standard receivers and decoders. A listing of the component make-up and weight breakdown is presented in Table 4.8-7.

Table 4.8-7 Plasma Flow Diagnostics Package Weight Breakdown

Item	Weight (LB)
Instruments	
Vector Magnetometer	9.0
Ion Mass and Distribution Analysis	4.4
Planar RPA	6.6
Langmuir Probe	7.7
Neutral Mass Spectrometer	22.0
Subsystems Equipment	
Communications	
Transmitter	1.1
Command Receiver	1.3
Diplexer	2.2
Antenna, Stub	2.2
Data	
Command Decoder	3.1
PCM Programmer	4.4
Subcarrier Oscillator	4.7
Power	
Power Supply	37.5
Cable Set	3.1
Thermal	
Strip Heaters	2.2
Multi-layer Insulation	7.7
Structures and Mechanisms	26.5
Total	147.5 (66.1 kg)

4.8.7 Summary of Maneuverable Subsatellite Trade-off

This section summarizes the results of a maneuverable subsatellite (MSS) analysis discussed in Section 5.9. The question addressed is: what are the advantages and disadvantages in moving the MSS up from Flight 4 of AMPS to Flight 1?

In Flights 1 and 2 there are five experiments (or parts of experiments) that can benefit from the use of an MSS. These are as follows:

- o Electron Beam Studies, Level II
- o EMI Field Mapping and Orbiter Wake Measurements
- o Conductivity Modification
- o Long-Delay Echo and Wave/Particle Interactions
- o Plasma Flow/Wake Generator

The use of an MSS in the above experiments can be shown to significantly enhance the scientific data over using a free flying package. The use of an MSS provides:

- o Greater data collection flexibility
- o More systematic measurement path
- o Greater number of measurements
- o More meaningful data with attitude control
- o Opportunity for more varied conditions
- o Measurements at greater distances from Shuttle
- o Opportunity for repeat measurements, and
- o Data storage alternative.

A list of candidate spacecraft is shown in Section 5.9. These include vehicles which are operational, under development and conceptual. There are some cost savings associated with the use of an MSS. These result from the elimination of flight support equipment (such as ejection mechanisms), instruments retrieved, and the elimination of the need for a second RMS. To offset the advantages given above NASA must provide early funding for an MSS. It remains to be determined whether it is more desirable to modify an existing vehicle or develop a new one.

4.9 AMPS Software System Requirements Definition

The general categories of AMPS operational software support are illustrated in Figure 4.9-1. Six elements of activity are identified, spanning the operational regime from mission planning to the collection and management of real time mission data with post-mission data analysis in mind. These general categories parallel the effort required on any space mission, but one aspect of the AMPS software development problem is unique. The AMPS operational software is not a stand-alone entity. Rather, its various parts are subdivisions of existing (or planned) software systems.

The AMPS software in Categories (2) and (4) of Figure 4.9-1 are packages within the software system being delivered for the Spacelab. Figure 4.9-2 shows the structure of the spacelab software system. Three AMPS support packages have been added to the Spacelab deliverable items to illustrate where the AMPS support software fits into the Spacelab software picture. These packages will be developed by the AMPS integrating contractor. They will be developed in accordance with Spacelab software packages design specifications to assure proper operation within the planned system.

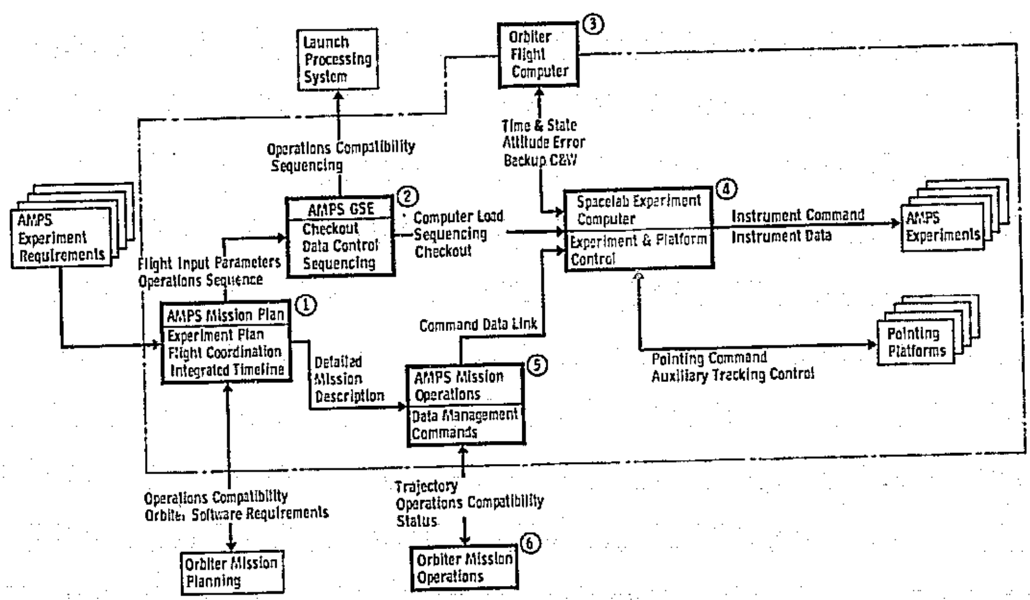


Figure 4.9-1 AMPS Software Support Function

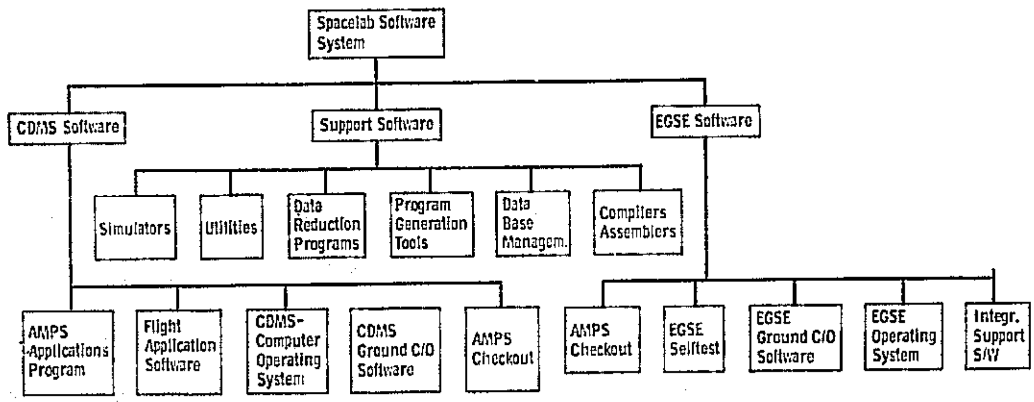


Figure 4.9-2 AMPS Supplement to Spacelab Software

In other areas, the relationships will be different. Consider Category (3) of Figure 4.9-1. The payload support computer in the Orbiter vehicle has some 10,000 words of core set aside for payload support. Certain elements of AMPS mission computational support that will require this support have been identified. This software will be developed by the Shuttle software contractors in response to requirements documents prepared by the AMPS integrating contractor.

A clear definition of functional requirements, and agency/contractor responsibilities in implementing solutions to these requirements is essential to the proper evolution of AMPS operational support software. The purpose of software analysis to date is: to provide a skeletal definition of these requirements; a categorization of their software implementation; and the associated contractor responsibilities for development.

4.9.1 AMPS Mission Planning

The overall purpose of the AMPS mission planning task is to evolve an integrated timeline of mission activity and to generate those media required to input this timeline into the appropriate operational elements. The general functions involved in mission planning are outlined in Table 4.9-1. This is the process whereby experiment objectives are played against the constraints of hardware/system/people limitations to come up with a realistic plan of what a particular AMPS flight will

accomplish. Many aspects of this process will not be mechanized in software, but other aspects can and will be enhanced by a software mechanization.

Table 4.9-1 AMPS Mission Planning Software

<table>
<tr><th>GENERAL FUNCTIONS</th><th>SOFTWARE ELEMENTS</th><th>SOFTWARE FUNCTION</th><th>DEVELOPMENT RESPONSIBILITY</th></tr>
<tr><td rowspan="6">o Evaluate Experiment Objectives/Requirements

o Integrate Mission Design Constraints
- Experiment/Instrument Imposed
- Orbiter System Derived
- Spacelab System Derived
- Communications Geometry
- Flight/Ground Crew Limitations

o Generate Detailed Mission Timeline
- Mission Orbit Sequence
- Orbiter Attitude Profile
- Experiment/Instrument Sequences
- Crew Operations Schedule
- Data Acquisition/Processing Requirements
- Command Generation Requirements
- Contingency Plans

o Prepare Input Media
- Coordination Reporting
- Training/Simulation Materials
- Computer Input Tapes</td><td>Preliminary Trajectory Design Package</td><td>Generate Specific Trajectory Maneuver Requirements for Orbiter Vehicle</td><td>Integrating Contractor</td></tr>
<tr><td>Primary Consumables Analysis Package</td><td>Generate Specific Propellant, Power, Thermal, Etc. Requirements</td><td>Integrating Contractor</td></tr>
<tr><td>Preliminary AMPS Timeline Package</td><td>Prepare Detailed AMPS Experiment Timelines</td><td>Integrating Contractor</td></tr>
<tr><td>Shuttle Mission Design Set</td><td>Develop Detailed Shuttle Mission Design</td><td>JSC</td></tr>
<tr><td>Spacelab Computer Utilities Packages</td><td>Prepare EGSE, Subsystem Experiment Computer Load Tapes</td><td>ESA</td></tr>
<tr><td>AMPS Equipment Simulation Packages</td><td>Provide Simulation Media For AMPS Integration and Checkout</td><td>Integrating Contractor</td></tr>
</table>

A preliminary listing of the software packages to be used in supporting the mission planning activity is also given in Table 4.9-1. With the exception of the Shuttle Mission Design package, all of these packages will be used by the integrating contractor at his central facility. They therefore comprise the "Mission Planning Software Set," according to the definition being used in the Spacelab Software Development Plan.

The preliminary trajectory design and consumables analysis software is the means whereby the AMPS integrating contractor integrates experiment objectives into a realistic mission plan. As the Orbiter/Spacelab mission is evolved, the details of the AMPS mission timeline are completed by the AMPS integrating contractor. Subsets of experiment and instrument timelines that form the individual elements of the AMPS mission are evolved by the responsible members of the scientific community, and are integrated into the total mission timeline and distributed for review by the mechanized process indicated. The mission plan is expected to encounter only minor changes during the detailed Shuttle mission design process that is carried out by JSC to prepare the complete input data set required to initiate a STS flight. Computer utilities evolved as a part of the Spacelab software system through ESA make the final bridge from mission planning to the operational computer hardware. Appropriate equipment output simulations are prepared as required to support the various levels of AMPS mission integration and checkout--leading to launch readiness on a timely basis.

The responsibilities for developing these software packages are shown in Table 4.9-1. JSC develops the STS mission planning system. ESA develops the basic Spacelab software system, which includes the input utilities. Both of these software elements will be available in advance of the AMPS support requirements. The remainder of this software is to be developed by the AMPS integration contractor. It should be noted that this set of software is less rigidly structured than the remainder of the software to be discussed in this development plan. It is not as time constrained in its operation, and more flexibility for alternate production procedures are possible. This projection, then, must be considered more in the line of a suggested concept than an operational requirement. Other approaches may be selected before the AMPS software becomes totally committed.

4.9.2 AMPS Payload Integration, Test and Launch

Preparation of the AMPS payload for launch requires some form of software support at each step of the process (Table 4.9-2). The initial steps--instrument development and Level IV integration--take place without benefit of the Spacelab EGSE and flight computers. Rather, a GFP computer is used to simulate the required interfaces. Level II integration, which physically mates AMPS and Spacelab, utilizes both the Spacelab EGSE and flight computers. In the final Level I integration, the Orbiter flight computer and the LPS computer systems are available, while the EGSE computer has been deleted from the operating configuration. As a consequence, the software configuration varies considerably as different objectives are met with the changing computer array. Note that the bulk of the software required is supplied by the Spacelab program. Only AMPS peculiar software needs to be developed by the AMPS integrating contractor.

Table 4.9-2 AMPS Payload Integration, Test and Launch Software

GENERAL FUNCTIONS	SOFTWARE ELEMENTS	SOFTWARE FUNCTION	DEVELOPMENT RESPONSIBILITY
o Support Instrument Development - Verify Instrument Control Techniques - Simulate Instrument Command/Data Interface o Support Level IV/III Integration - Verify Flight Application Modules - Simulate Spacelab Functional I/F - Simulate Orbiter Functional I/F - Provide Test Stimulus/Response Verification o Support Level II Integration - Simulate Orbiter Functional I/F - Provide Test Stimulus/Response Verification o Support Level I Integration - Provide LPS Interface	Spacelab Provided Sets - EGSE Checkout - Electrical System Integration - Ground Checkout - Software Simulation* - Data Reduction* - Flight	This represents the complete Spacelab operational software system which will be provided for payload integration	ESA
	AMPS - EGSE, Checkout Package	Provides Stimulus Generation--Response Monitoring Required To Check Out AMPS Payload	Integrating Contractor
	AMPS - CDMS Checkout Package	Provides CDMS-Resident Software Required To Interface With EGSE	Integrating Contractor
	AMPS - Orbiter Support Package	Provides Orbiter Flight Computational Support For AMPS Experiments	JSC
	AMPS Flight Applications Packages	Provides Airborne Monitor and Control For AMPS Experiments	Integrating Contractor
	AMPS/LPS Interface	Provides Data Required For LPS Monitor Of AMPS Status	Integrating Contractor

*Also Included In Software Development Support Set

The initial instrument integration task is conducted by the instrument contractor at his home location. It is important, at this stage, for the instrument contractor to be assured that his data output is compatible with the AMPS system, that his operation sequences can be accommodated, and that his instrument is compatible with the commands issued by the AMPS software system. These objectives are met by simulations implemented in a computer located at the integrating contractor's site and relayed in a quasi real-time mode via telephone.

The initial AMPS payload integration, Level IV, takes place at the integrating contractor's facility. This integration is completely supported by simulations of the operating environment created on the host computer. The basic software required is provided from the Spacelab program. The only additions required are the various AMPS applications packages that are required for mission operations. Level III integration is functionally similar to the Level IV integration. The difference is that it will be conducted at KSC with their complete Spacelab simulator. The software developed for the Level IV integration is portable, and required to be compatible with the host computer at the integrating contractor's site and the host computer at KSC.

Level II integration mates the AMPS payload with the Spacelab flight equipment and electrical ground support equipment. At this point, a completely realistic flight environment for the AMPS instruments is achieved, although the Spacelab/Orbiter interface is still simulated. A complete stimulus/response checkout of instrument operation is effected using AMPS applications packages in both the Spacelab flight computers and EGSE computer. On completion of this sequence of tests, the AMPS payload is ready for flight. Only health monitoring is required from this point forward. It is effected in a routine manner by the flight applications packages and reported as a go/no-go situation.

Thus Level I integration for the AMPS payload is no different from the flight situation. Mission control response to an out of limits situation may vary, but the software required is largely the same. The only difference identified is the software link that reports any AMPS hazards to the launch processing system. While not known at this time, this link is expected to be implemented by tabular inputs to Spacelab supplied software.

The development responsibilities for the identified AMPS payload integration, test and launch software are indicated in Table 4.9-2. ESA will supply the bulk of the Spacelab and Spacelab/Outside World simulation software. JSC will provide the Orbiter AMPS payload support package (in response to AMPS generated requirements). The AMPS integrating contractor will supply the appropriate AMPS application packages.

4.9.3 Orbiter Airborne AMPS Support

The Orbiter provides AMPS experiments with their primary positioning/pointing, navigation/time references, and communication links with the

ground (Table 4.9-3). Requirements also exist for the Orbiter to track and determine the orbits of subsatellites, and to maneuver instruments around the vehicle with the RMS. Most of these requirements are met by currently planned Orbiter capabilities. However, some require AMPS peculiar extensions of capability.

Table 4.9-3 Shuttle Airborne AMPS Support

GENERAL FUNCTIONS	ORBITER P/L SUPPORT PACKAGE	SOFTWARE FUNCTION	DEVELOPMENT RESPONSIBILITY
Standard Payload Services o Reference Data - Time - Navigation State - Attitude - Subsatellite o Data Handling - Backup Caution & Warning - Command Uplink - Data Downlink o Attitude Control - Desired Attitude - AMPS Attitude Error	Standard Support Modules - Orbiter State Vector o Position o Velocity o Attitude o GMT/MET - Target State Vector - Payload Attitude - Limit Check - Command Management - Telemetry Management	- Experiment Tag Data - Subsatellite Orbiter Parameters - Attitude Control by AMPS Sensors - Back Up Caution and Warning - Ground Command Uplink - Telemetry Downlink	JSC
	AMPS Peculiar Modules - RMS Maneuver Program - RMS State Output	- Precise EMI/Electron Beam (Et. Al.) Mapping - Experiment Tag Data	JSC
AMPS Peculiar Services o RMS Maneuver Program o RMS State Output	Tabular Input Parameters - Command Schedules - RMS Maneuver Requirements - Limit Ck Data	Input Data Required to Meet AMPS Flight Requirements	Integrating Contractor

The primary positioning and pointing is introduced as an integral part of the Orbiter mission plan. A capability is required, and is provided by the standard Orbiter payload support software, to close the Orbiter attitude control loop with a payload generated attitude error. This capability is generated as an AMPS response to an Orbiter request for payload attitude data.

Orbiter state data, required for AMPS pointing platform control and AMPS experiment data logging, is also provided by the standard Orbiter payload support software. Transmission of this data to AMPS is scheduled in the basic Orbiter mission plan, either under software or crew control.

Standard provisions have been made in the Orbiter payload support software to transmit the Orbiter state data (position, velocity and attitude, as well as time) required by AMPS. A provision also exists to transmit target state vector data. This capability will be used to transmit subsatellite orbit determinations to AMPS.

Data required by the Orbiter to assure the safe operation of the AMPS payload is entered into the PCM data stream for limit checking and display as backup caution and warning parameters. This is a standard Orbiter capability. Relay of this data to the ground, and handling of ground uplink commands are also standard provisions.

Non-standard services are required for RMS usage. The AMPS experiments require precise, complex movements from the RMS. Further, position and attitudes achieved by the RMS are required by AMPS. Automation of RMS movement schedules, and feedback of RMS state data require that the new modules indicated in Table 4.9-3 be added to the Orbiter payload support software.

The standard elements of Orbiter payload support software are being developed by JSC, as indicated in Table 4.9-3. The new elements of payload support software will also be developed by JSC, in response to requirements documents generated by and subject to functional verification by the AMPS integrating contractor. This payload support software is generally controlled by tabular input data. These data are to be generated by the integrating contractor, subject to verification by JSC during the Shuttle mission design activity.

4.9.4 AMPS Operational Flight Support

The operational flight software provides the means whereby the flight crew directs the conduct of the AMPS experiments, and provides the data time-tagging that permits post-flight reconstruction of data collection events. The support required includes tutorial information, instrument monitor and control, processing decision-influencing scientific data, and providing that all data collected is properly preserved for post flight evaluation. The design of this software is driven by the desire to achieve the maximum scientific usefulness of the AMPS missions. It provides the flight crew with the information and control capabilities required to use their judgment to maximize the value of the scientific data output.

The general software functions indicated in Table 4.9-4 reflect a level of automation designed to relieve the flight crew of non-productive tasks. A convenient interface with the experiments is made through function keyboards and displays. Automated timeline service is provided, with a capability for the flight (or ground) crew to alter the timeline in response to real time events. Routine monitoring functions are handled automatically, with warnings and diagnostics provided in the event of system failures or unanticipated events. Routine scientific data handling is done automatically. This approach allows maximum crew attention to be given to observing and maximizing the quality and optimizing the volume of scientific data yield.

Table 4.9-4 AMPS Operational Flight Support Software

GENERAL FUNCTIONS	SOFTWARE ELEMENTS	SOFTWARE FUNCTION	DEVELOPMENT RESPONSIBILITY
o Provide Crew/Experiment Interface - Interface Aids - Command Interface - Display	SCOS/ECOS Packages (Subsystem & Exp Computers)	Manage Computer Resources, Input/Output, Displays & Data Links	ESA
o Provide Automated Timeline Service - Tutorial Information - Schedule Support - Detailed Sequencing	Inflight Subsystem Checkout Packages (Subsystem & Exp Computers)	Perform Checkout & Fault Isolation of Subsystems, Display Status & Warning Messages to Crew	ESA
o Maintain Experiment Health Status - Limit Checks - Warning - Diagnostics/Tracing	Inflight Monitor Packages (Subsystem & Exp Computers)	Periodically Monitor Selected List of Parameters, Compare to Limits & Issue Resultant Messages	ESA
o Provide Real Time Data Handling - Data Tag - Routing - Processing	Power & Energy Management Package (Subsystem Computer)	Provide Present Level of Power Usage and Mission Energy Consumption	ESA
o Support Auxiliary Experiment Point/Maneuver Requirements - Pointing Platform Management - Orbiter Attitude Error Computation - Subsatellite Control - RMS Motion Requirements o Provide Software Executive Control - Computer Time Sharing - Module Control - Command Interpretation	Experiment Packages - Display Generation (Exp) - Command Interpreter (Exp) - Operation Data Base (Exp) - Downlink Config (Exp) - Pointing Reqmts (Exp) - Point Platform Mgt (Subs) - RMS Reqmts (Exp) - Sub Sat Mgt (Exp)	Provide Experiment Displays Interpret Keyboard Commands Provide Tutor, Sequences, Et. Al. Configure Telemetry Formats Establish Experiment Point Reqmts Direct Experiment Point Platforms Establish RMS Maneuver Sequences Target, Release & Predict Sub Sats	Integrating Contractor

Analyses run to date indicate that the total quantity of software required to support one scientific mission is likely to grow larger than the capacity of the Spacelab airborne computers. Therefore, a capability of transferring applications modules from mass memory to computer core is required. Only these elements required to support a flight segment are resident in the computer at any one time.

The basic management/data handling capabilities required on the AMPS mission are supplied by the Spacelab Computer Operating Systems (ECOS for the experiment computer, SCOS for the subsystem computer). CPU time sharing, reading applications modules in from the mass memory, keyboard and display interfaces, and data handling through the remote acquisition units are all controlled by the operating systems. This software package is supplied by ESA as a part of the Spacelab software system. Its capabilities and interfaces are established by ESA and form a constraint on the design of AMPS application packages.

Two of the Spacelab supplied applications packages noted in Table 4.9-4 are of specific use in the AMPS application. The inflight monitor package in the experiment computer provides an automated means for maintaining a status check on the health of the AMPS instruments. The power and energy management package provides a means of keeping track of the AMPS power usage plan.

Command of the AMPS experiments and display of mission progress will be a unique applications package--designed to meet the specific needs of this program. At the present time, an efficient interactive approach is planned. The approach uses displays that are keyed to particular experiment/instrument characteristics, and function keyboard assignments that dovetail with the display design. The particular approach will be developed by the AMPS integrating contractor, in conjunction with the various experimenters and instrument suppliers involved.

An operations data base will be supplied for use by the flight crew. This data base will include tutorial information, automated instrument operation sequences, and guidelines relating to (and/or overall scheduling of) experiment operations during individual AMPS flights. The overall flight operation will be under the direct control of the flight/ground crews, but the maximum useful automated operational aids will be provided. One aspect of this support will be automated configuration of the downlink telemetry stream to support the particular experiments being conducted in the current timeframe.

A particular sharing of the duties associated with pointing platform management and control has been assumed--other arrangements are possible, but must be evaluated carefully before changing the philosophy. The experiment computer is charged with establishing the time history of platform pointing in an Earth centered (or other appropriate) coordinate system that is compatible with experiment requirements. The subsystem computer is charged with transforming these requirements into platform gimbal commands and modifying these commands using auxiliary data sensed at the platform (e.g., star trackers). High data rate platform stabilization, using inertial sensors on the platform, is accomplished by auxiliary stabilization devices that are a part of the platform system--and do not complicate the data interface with the Spacelab computers.

Instruments are placed in remote locations in accordance with AMPS requirements by the Orbiter RMS, and by subsatellites that are a part of the AMPS payload complement. A provision to generate and transmit RMS motion requirements to the Orbiter is provided. Also provision to generate subsatellite release requirements (time and orientation), and to predict subsatellite motion based on release characteristics and/or Orbiter tracking data is provided.

All of these AMPS peculiar applications modules are used in the Spacelab experiment computer. Their development is the responsibility of the integrating contractor. Their design must be compatible with the Spacelab operating systems that will control and schedule their operation. The Spacelab packages indicated in Table 4.9-4 are the development responsibility of ESA. The complete system will meet all AMPS mission requirements.

4.9.5 Payload Operation Control Center Support

The POCC is charged with the responsibility of participating with the flight crew in maximizing the scientific return of each AMPS flight. It is also responsible for collecting, retaining and distributing this scientific return to the community for which it was gathered. These objectives are complementary to the Mission Control Center objective of safely and efficiently conducting the flight, subject to realistic constraints imposed by STS considerations.

The first part of this task is oriented towards influencing the conduct of the flight as it is taking place. This requires real time display of critical portions of the downlink data stream, quick look analysis of the progress of the mission, and real time replanning of the mission in response to the situation encountered (see Table 4.9-5). A carefully thought out approach to the software design supporting the crew interface with mission events is required to meet this objective.

Table 4.9-5 Payload Operations Control Center Support Software

GENERAL FUNCTIONS	SOFTWARE ELEMENTS	SOFTWARE FUNCTION	DEVELOPMENT RESPONSIBILITY
o Data Decom/Distribution/Display - Accept & Assemble AMPS Data - Distribute Real Time, Retention Data - Display Process Real Time Support Data o Quick Look Analysis - Data Quality - Instrument Evaluation - Scientific Conclusions o Contingency Planning - Malfunction Workarounds - Scientific Content Maximation - Rescheduling/Optimization o Command Generation-Uplink Flight Modification o Data Archival-Permanent Retention o Scientific Data Processing - Data Conversions - Group by Experiment - Data Reports o Post Flight Analysis - Malfunction Analysis - Objective Evaluation - Future Plans Modification	<u>Realtime Support</u> - Executive - Telemetry Processing - Data Records Generation - Language Processor - Sequence Editor - Command Generation - Applications Modules	 Equipment Mat'l & Process, Schedules Downlink Data Handling Recording & Archiving Operator Dialog Interface Timeline Modification Processor Generate Uplink Formats Experiment/Instrument Dedicated	GSFC
	<u>Off Line Support</u> <u>TBD</u>	Analysis Tools Requirements to Support Specific Experiment and Instrument Objectives	Integrating Contractor

The other aspect of this task is the preparation and distribution of the scientific data gathered. All data gathered must be retained in its original state--the retention of only processed data makes it impossible to recover from any inadvertent processing error. Then, data reports providing each investigator with all the data he requires without burdening him with extraneous information are required. This latter

task requires careful planning for maximum effectiveness. A considerable body of well planned software is required to meet this objective on a timely basis.

The conceptual structure of the real time support software required to meet POCC objectives is shown in Table 4.9-5. All downlink data handling and uplink commanding is automatically effected by appropriate software. The operator interface is a critical aspect of this problem. A convenient user language will be required to give the operator flexible control of data handling, display and processing. Another significant aspect of this interface is the editing tools used to modify the automatic sequences governing experiment conduct. Finally, provisions for experiment/instrument dependent applications modules must permit special support to be quickly and easily introduced into the real time data processing system.

Off line support to the POCC will be where most of the experiment/instrument specific software will reside. This element is not well understood at this time, and will evolve slowly as AMPS flight definitions become firm, and probable operational problems become better defined.

It is anticipated that the real time software will be derived largely from existing GSFC operational packages. These elements will require redesign to fit the AMPS mission concepts, where a flight crew exists to support the overall objectives. This software development, therefore, will probably be largely a GSFC responsibility, with efforts subcontracted as they see fit. Off-line support will probably be most readily defined and implemented by the integrating contractor.

4.9.6 STS Mission Control Center Support

STS mission control is primarily concerned with the considerable problems associated with putting the AMPS payload safely on location. Maximizing the quality of the scientific return, on the other hand, is the problem of the payload operations center. These objectives are compatible but must remain within constraints imposed by the STS. The relationship has to be that payload objectives will be made known to the MCC, and MCC will respond within its capability. This relationship minimizes the detailed AMPS information that must be made available for real time decisions at the MCC. Further, there is a minimal requirement there for archiving/distributing these data.

Three categories of AMPS data are required in the MCC to meet these objectives--AMPS related safety data, proposed AMPS flight profile modifications, and AMPS command sequences required to effect the profile modifications. The required safety data is made available to the Orbiter crew in-flight through the PCM telemetry data stream. The same data is readily available at the MCC. Proposed flight profile modifications are generated in the POCC. These data must be reviewed at the MCC to assure that they do not violate STS constraints. After approval, the command

sequence affecting them are transmitted from the POCC to the MCC for entry into the STS command uplink.

No AMPS unique software packages have been identified for use in the STS mission control center. Rather, it is required that the MCC software be developed with AMPS interface requirements in mind. A capability of considering AMPS data must be provided and appropriate flexibility in the design of the MCC software is required. These requirements must be identified in AMPS requirements documentation, and implemented in JSC's design of the STS mission control software.

4.9.7 Software Development Support

This category includes software that is required to create and simulate the AMPS operational software. It is a necessary part of the software system that must be written, checked out and used--but it does not form a direct part of the operational mission. All required aspects of this software set are being developed by ESA, and will be available for production of the AMPS-peculiar software elements.

The software production set is delineated in Table 4.9-6. Compilers are available for conversion of higher order languages (HAL/S, FORTRAN, GOAL) to assembly language for the 370 and Mitra 125 S/MS computers. Machine peculiar macro assemblers and linkage editors create binary code suitable for loading into the host and flight computers. Various utilities provide a means to create and maintain libraries, data bases, etc.

Table 4.9-6 Software Development Support Software

GENERAL FUNCTIONS	SOFTWARE ELEMENTS	SOFTWARE FUNCTION	DEVELOPMENT RESPONSIBILITY
o Software Production	S/W Production Set		
- Compile/Assemble/Checkout/ Integrate	- Automated Management	Library Management	
- Load Module Production	- Set Integration	S/W Production Utilities	
- Data Base Generation & Maintenance	- Data Base Generation	Checkout Data Base	
	- Macro Assemblor (Host)	Generate Relocatable Binary Code	
o Data Handling	- Linkage Editor (Host)	Produce Loadable Machine Code	
- Checkout, ESI, Simulation & Flight Data Conversion	- HAL/S 360 Compiler	Compiles HAL/S for 370	
- Data Analysis/Correlation	- HAL/S CII Compiler	Compiles HAL/S for MITRA RSS/MS	
	- Checkout Language Compiler	Creates Input for Language Interpreter	
o Simulation	- MACRO Assembler (CII)	Generate Relocatable Binary Code	
- Interpretive Computer Simulation	- Linkage Editor (CII)	Generate Loadable Machine Code	
- Peripheral Equipment Simulation	- ANSI/FORTRAN Compiler	Generate Relocatable Binary Code	
- Interface Simulation	- Miscellaneous Utilities	Utilities for S/W Production On 125 S/MS	
	Data Handling		ESA
	- EGSE Data Reduction	Produce Test Results Reports	
	- Host Data Reduction	Produce Test Result Reports	
	- PCM Analog Tape to Digital Tape	Data Conversion	
	Simulation		
	- I/O Box & Peripheral Sim	Simulate CDMS Hardware	
	- EGSE Simulator	Simulate EGSE Reactions	
	- Spacelab Simulator Coord	Simulate Spacelab Subsystems	
	- Interpretive Computer Sim	Simulate MITRA 125 S/MS In 370	

Data reduction software provides an automated means of processing log tape data or filed data and producing test result reports. Methodology shall encompass checkout, integration and flight data. Capability for trend analyses based on several tests or flights shall be provided. Implementation shall be in ground support computers, as opposed to the flight computers.

Simulation software shall be used in software development, and the various steps of flight/experiment integration. The interpretive computer simulation provides a means to check out EGSE/flight code in the IBM 370 host computer, with the added benefit of various diagnostic capabilities such as TRACE/JUMP TRACE/SNAP. Other simulations used in the various phases of payload integration (before the actual flight hardware can be mated) includes simulation of CDMS peripherals, EGSE equipment, and Spacelab/Orbiter interfaces. The use of these simulations maximizes the probability of each succeeding integration step proceeding without incident.

5. SUPPORTING ANALYSES

5.1 Systems Level Analyses

5.1.1 AMPS Flight 1 Preliminary Trajectory Design

This section presents the results of a trajectory design analysis of the first AMPS flight. Sunlight/shadow data, maneuver schedules and maneuver propellant requirements are included. While these data are preliminary in nature, they do prove the feasibility of the design concepts utilized.

Trajectory Design Objectives - The first AMPS mission is inclined 57° to the equator, is conducted at altitudes in the neighborhood of 205 kilometers, and is oriented to produce a sunlit vehicle during predawn passes over Aercibo, Puerto Rico on the first three days of the seven day mission. After the first three days, passes over Arecibo continue, but the condition of a sunlit vehicle is not maintained.

The 57° inclination results from the desire of the Minor Constituents and other first flight AMPS experiments to possess the highest inclination available within the normal Shuttle ascent range from KSC. The approximate 205 km altitude, and the sun/shadow relationships result from the desire of the Gravity Wave experiment to have a capsule gas release occur in sunlight while visible to optical instruments located at Arecibo. Higher repeating orbits are generally inconsistent with AMPS experiment objectives. These requirements dictate the placement geometry of the mission orbit: a launch time selected to produce the desired lighting, an altitude selection permitting daily passes over Arecibo, and maneuvers to achieve and maintain this daily overflight.

Trajectory Design Technique - The AMPS trajectory design has been initiated at the time of external tank ejection on a 35° azimuth Shuttle launch from KSC. The position at this time is Lat. 38.2°N, Long. 288.7°E, Alt 116 km, approximately 490 seconds after liftoff. The orbit is 36 km x 158 km, which results in external tank impact in the Indian Ocean. A burn is immediately required to raise this apogee to a desirable operating altitude and a burn is required to circularize the orbit when this altitude is reached. In this trajectory design, these first two burns use the Orbiter OMS engines, while subsequent burns use the RCS engines. No out-of-plane propulsive maneuvers are planned; rather all trajectory design objectives are met by orbit period adjustments that time orbital passes such that the desired ground features pass through the orbital plane at the correct time.

The first day's objective is to bring the orbiting vehicle directly over Arecibo on the 15th orbital pass as shown in Figure 5.1.1-1. This is accomplished by maneuvering the orbiter to an appropriate altitude, maintaining orbit altitude with a drag makeup maneuver, and then bringing the Orbiter down to a lower operating altitude for a pass over

Arecibo. The higher operating altitude during the first day's operations is required to slow the orbital period slightly so that Arecibo can rotate into the orbital plane on the 15th pass. Note that a 16-orbit daily repeating pattern is maintained from the second day onward. Launch time is selected to achieve the desired sun/shadow relationship.

The only maneuver objective on the second day is to maintain the orbital pass over Arecibo. Two drag makeup sequences are scheduled to achieve this objective. On the third day, a single drag makeup sequence is employed. The selection of one and two drag makeup sequences per day was arbitrarily selected for purposes of comparison.

Trajectory Design Summary - The drag attitudes used in this trajectory simulation reflect anticipated vehicle attitudes required to perform AMPS experiments during the first flight. These attitudes, and the resultant vehicle area presented to the velocity vector, are presented in Table 5.1.1-1. A drag coefficient of C_D = 2.2 was used for all attitudes.

Table 5.1.1-1 Drag Attitude History

Time From Liftoff Start (Hrs)	End (Hrs)	Attitude	Exposed Area (Sq Meters)
0	6	Front Exposed	45.0
6	10	Profile Exposed	185.9
10	11	Base Exposed	365.3
11	17	Profile Exposed	185.9
17	20	Base Exposed	365.3
20	34	Profile Exposed	185.9
34	35	Base Exposed	365.3
35	41	Profile Exposed	185.9
41	44	Base Exposed	365.3
44	58	Profile Exposed	185.9
58	59	Base Exposed	365.3
59	65	Profile Exposed	185.9
65	68	Base Exposed	365.3
68	82	Profile Exposed	185.9

Figures 5.1.1-1, 5.1.1-2 and 5.1.1-3 show the first three days ground traces, each of which was arbitrarily terminated over Arecibo, the ground station used to monitor the gravity wave experiment. The points where the vehicle enters and emerges from the earth's shadow are indicated. Note that emergence from the shadow is approximately 10° before arrival at Arecibo on the first day, reduced to approximately 3° before arrival on the third day. This effect is due to a combination of orbital regression (due to oblateness) and the earth's motion around the sun causing the orbit to regress into the dawn shadow. This effect is also shown in Table 5.1.1-2, where the time of arrival at Arecibo is about 23 minutes earlier each day.

Orbital maneuver locations are marked on the orbital traces of Figures 5.1.1-1, 5.1.1-2, and 5.1.1-3. Timing and propellant required by these maneuvers is summarized in Table 5.1.1-2. The first day's maneuvers required 2875.5 kg of OMS propellant and 1069.6 kg of RCS propellant. It is evident that most of these maneuvers could have been done with the OMS, and probably will be in the final mission design. The large propellant requirements on the first day are mainly associated with ascent to operating altitude, and phasing to reach the desired pass directly over Arecibo. They are not representative of drag makeup requirements.

The second day employed two drag makeup maneuvers, while the third day was set up with only one. The second day required 434.5 kg of propellant, while the third day required only 172.0 kg of propellant. The inconsistency of these data led to an investigation of the accuracy of the trajectory integration techniques used. The results indicated significant variations between techniques. The conclusion is that the drag makeup requirements derived here are only approximately correct, and should be investigated in more depth at a later date. Since drag makeup requirements at these altitudes is a significant function of the time varying atmospheric density (a facet not considered in this study), a thorough investigation should include this effect as well as analysis of the accuracy of trajectory integration technique.

<u>Conclusions</u> - The trajectory analyses reported here have demonstrated that the Flight 1 design concepts developed during the study are feasible, and that the results reported are essentially correct.

The trajectory design evolved here is a workable approach, but it is possible to improve on it by reducing the number of propulsive maneuvers. It is also possible to change the times at which these maneuvers occur, if it should prove advantageous to the conduct of the AMPS experiments.

The drag makeup results obtained are questionable, and this area should be the subject of a future, detailed investigation.

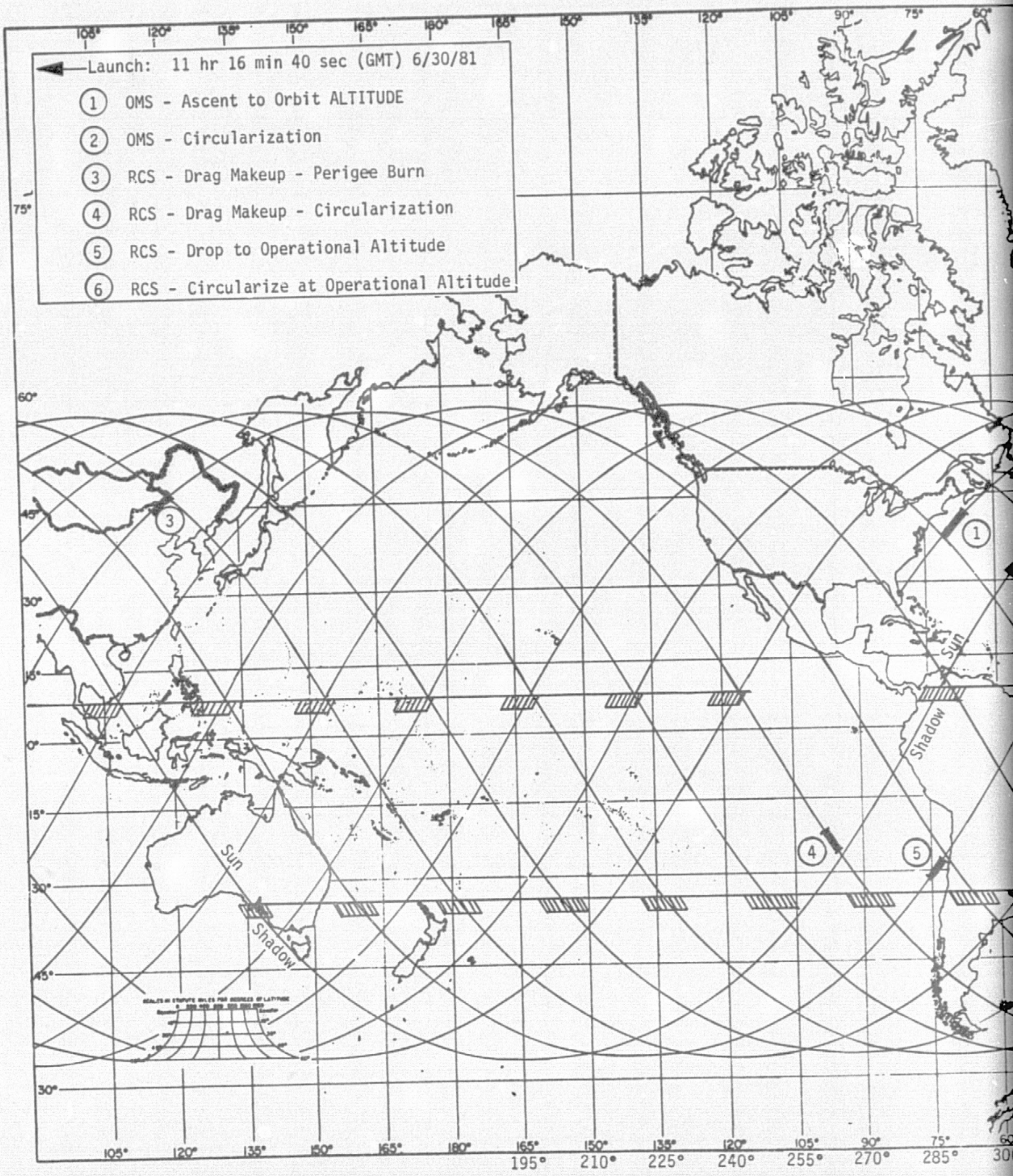


Figure 5.1.1-1 AMPS Flight 1 - D

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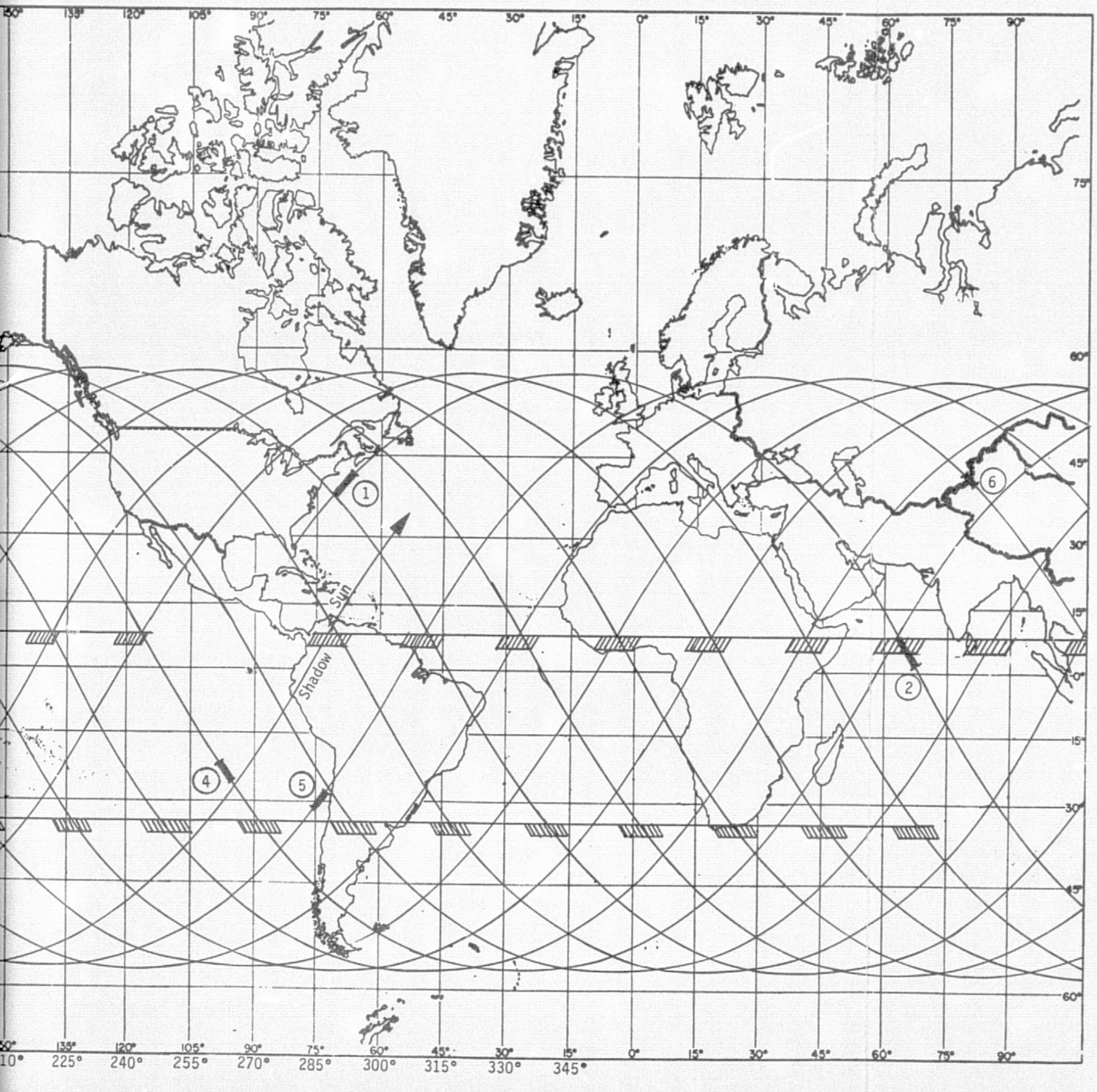

igure 5.1.1-1 AMPS Flight 1 - Day 1 Ground Trace

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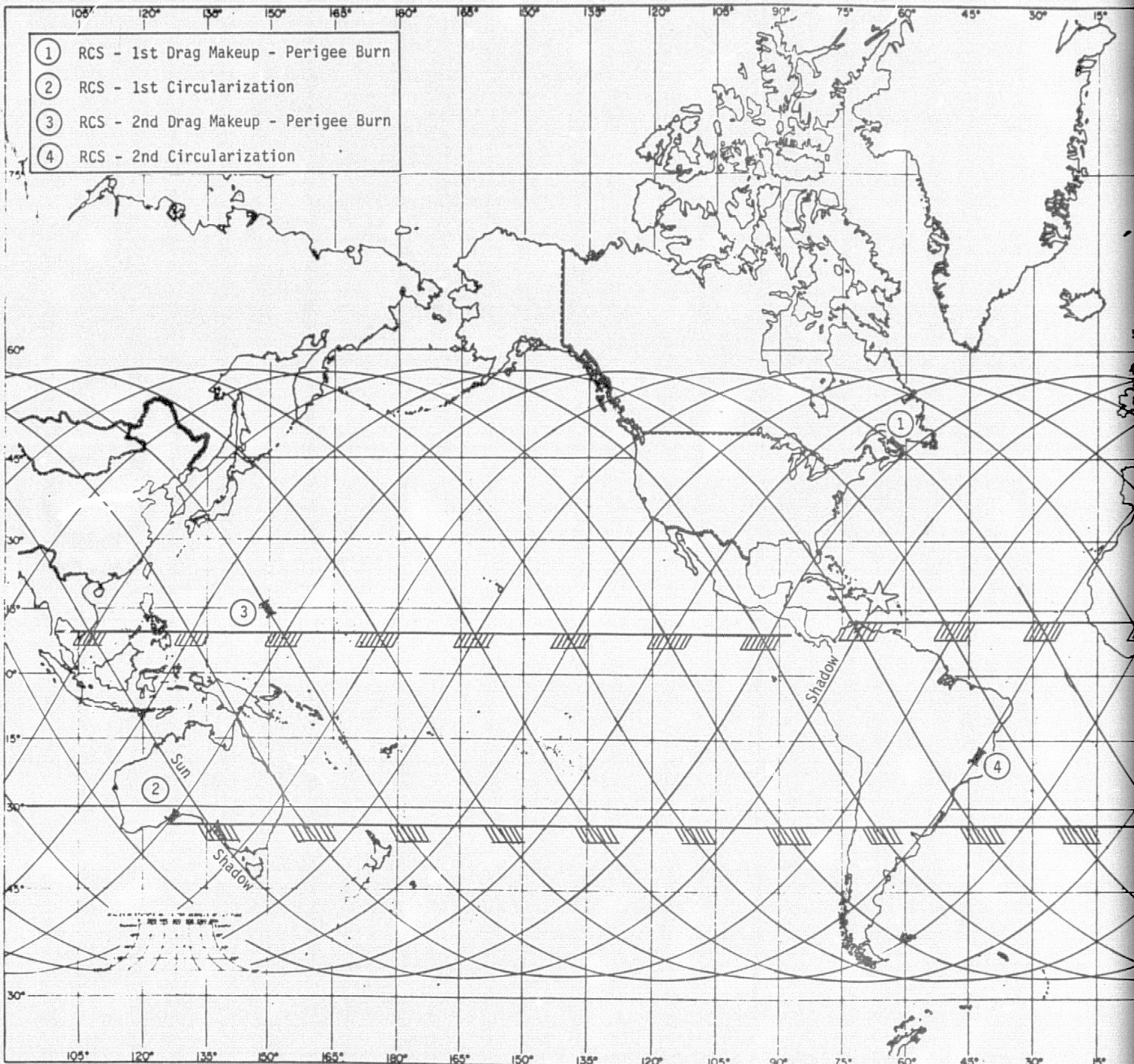

Figure 5.1.1-2 AMPS Flight 1 - Day 2 Ground Trace

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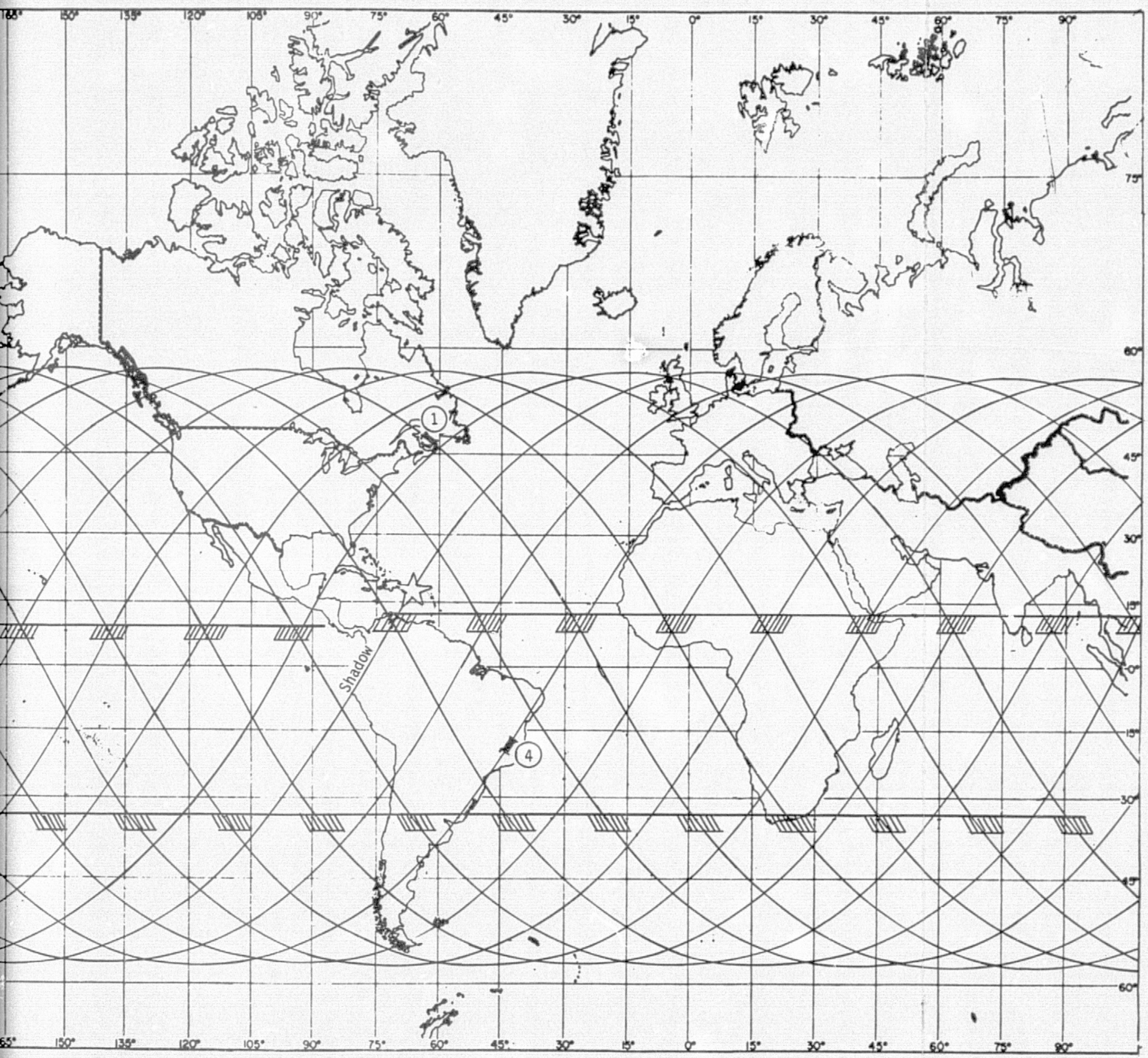

Figure 5.1.1-2 AMPS Flight 1 - Day 2 Ground Trace

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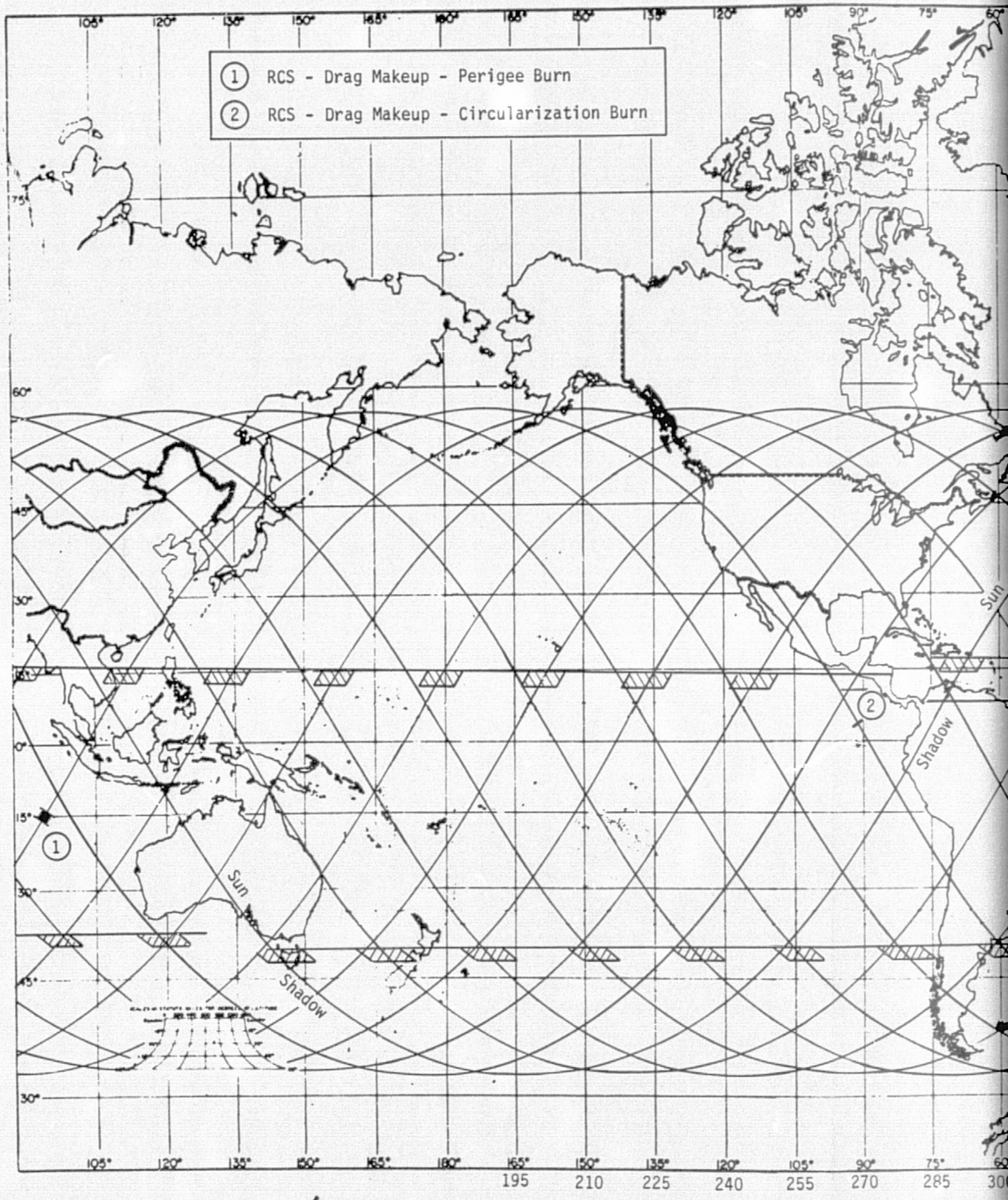

Figure 5.1.1-3 AMPS Flight 1 - Da

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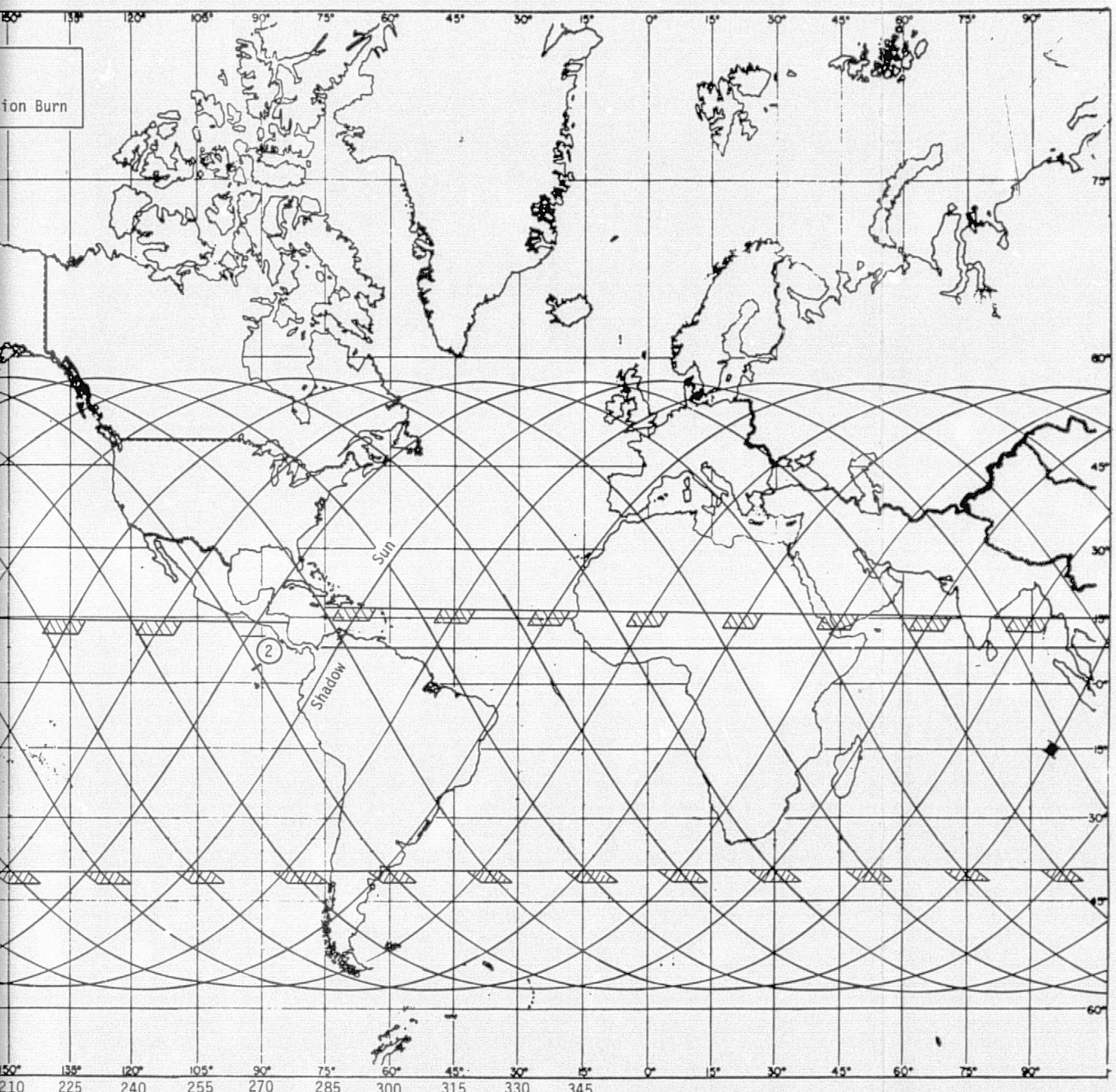

igure 5.1.1-3 AMPS Flight 1 - Day 3 Ground Trace

Table 5.1.1-2 AMPS Flight 1 Maneuver Summary

DAY	PROPULSIVE MANEUVER	PROPULSION SYSTEM	START TIME (FROM LAUNCH) (HR:MIN:SEC)	BURN DURATION (SEC)	PROPELLANT USED (kg)	CUM PROPELLANT (kg) OMS	CUM PROPELLANT (kg) RCS	ALT AT START MANEUVER (km)	ALT AT ARECIBO (km)	DAILY MAX ALT (km)	DAILY MIN ALT (km)	GMT
1	Launch		0:00:00.0									11:16:40.0 (6/30/81)
	Ascent Inject Burn	OMS	0:06:09.4	83.5	1451.3	1451.3	0	115.7				
	Circularization Burn	OMS	0:38:36.4	82.0	1424.2	2875.5	0	221.7				
	Drag Makeup - PGE Burn	RCS	12:00:00.0	54.0	295.2	2875.5	295.0	220.5				
	Drag Makeup - Circularization	RCS	12:37:1[illegible].9	49.3	269.2	2875.5	564.2	241.0				
	Maneuver to Operational Alt	RCS	20:31:38.4	48.9	267.2	2875.5	831.4	232.9				
	Circularize at Op. Altitude	RCS	21:11:30.6	43.6	238.2	2875.5	1069.6	211.8				(7/1/81)
	Arrive at Arecibo		22:14:30.6						207.1	250.8	115.7	9:31:10.6
2	Drag Makeup - PGE Burn	RCS	30:00:00.0	23.3	127.4	2875.5	1197.0	212.2				
	Drag Makeup - Circularization	RCS	30:50:13.8	26.4	144.3	2875.5	1341.3	220.3				
	Drag Makeup - PGE Burn	RCS	42:00:00.0	23.3	127.4	2875.5	1468.7	205.8				
	Drag Makeup - Circularization	RCS	42:43:43.0	6.5	35.4	2875.5	1504.1	215.0				(7/2/81)
	Arrive at Arecibo		45:51:36.4						208.7	226.9	201.1	9:08:16.4
3	Drag Makeup - PGE Burn	RCS	57:30:00.0	30.4	166.2	2875.5	1670.3	209.8				
	Drag Makeup - Circularization	RCS	58:18:00.6	1.1	5.8	2875.5	1676.1	216.8				(7/3/81)
	Arrive at Arecibo		69:28:42.7						198.1	225.7	196.2	8:45:22.7

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5.1.2 GSE/Facility Systems Level Analysis

The AMPS Phase B Ground Support Equipment and Facility Analysis was initiated by the preparation of a functional flow diagram shown in Figure 5.1.2-1 which presents an overview to the method of performing the task. The analysis began by taking program documentation as inputs for performing a requirements analysis. In the requirements analysis we divided the work into tasks which enabled us to define GSE and Facilities Requirements to perform the AMPS program. By comparing these requirements to existing or planned GSE/Facilities we were able to define the GSE/Facilities required to support the AMPS Phase C/D project along with the supplier of each item. This process has been completed as far as the program definition is known to date, but as noted in Figure 5.1.2-1 the process must be iterated to a more detailed level, as the Phase C/D program matures. The input phase of the program was started by a review of Shuttle, Spacelab and AMPS documents. The documents which were reviewed included the Space Shuttle Accommodations Handbook, the Spacelab Accommodations Handbook, the Launch Site Accommodations Handbook, the Spacelab GSE Items Description and Allocations Documents, the AMPS Instrument Functional Requirements Documents (IFRDs) and the AMPS Experiment Operations Requirements (EORs).

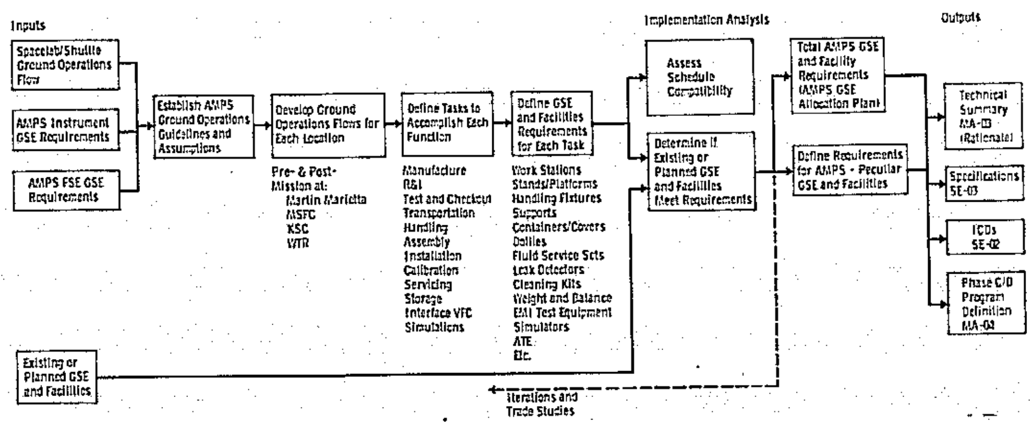


Figure 5.1.2-1 GSE/Facility Analysis Functional Flow Diagram

Following the documentation review, the requirements analysis phase began by generating GSE/Facility groundrules, which were used as an aid for developing future tasks.

- Design, development, test, transport, support and handling GSE for instruments and FSE will be used as applicable throughout the ground operations cycle, and, wherever required GSE built by the developing contractor will be delivered to support the planned activity.

- GSE identified as MMSE or commercial equipment will be used to support AMPS testing at all levels in preference to developing special GSE.

- GSE for transportation and handling of pallets and racks will be provided by Spacelab or MMSE.

- GSE must support development, test, transport, storage, launch preparation activities, both online and off, and during maintenance and refurbishment activities.

- The AMPS prime contractor will provide GSE (not available from the developer) required to support Level IV and subsequent activities.

- Existing facilities will be used wherever possible. Prime Contractor Facilities will be used for Level IV activities.

- No special handling or support equipment will be provided by AMPS for alternate site landing.

- No special handling or support equipment will be provided for post flight operations to remove film or magnetic tape from the payload prior to landing +12 hours (i.e., normal vehicle access in the Orbiter Processing Facility (OPF)).

- GSE design will be compatible with the planned AMPS payload evolutionary approach and as such shall not require redesign and build between flights but will require only necessary update modifications.

- GSE required for integration activities will be designed for use in a clean room environment.

- GSE which is shipped between facilities with the FSE or instruments shall be cleaned and bagged prior to movement.

- Access GSE from the Payload Changeout Room to the AMPS/Spacelab Payload interface connections to support AMPS unique payload activity will be provided GFP from KSC. Unique payload

GSE required to support the instruments or FSE during this time will be provided by AMPS Prime Contractor.

- The OPF will provide payload handling GSE necessary to support all AMPS payload requirements in that facility (i.e., Hoist capabilities to 65,000 lbs (29,483.5 kg) with a 15 ft (4.57 m) diameter and a 60 ft (18.3 m) length.)

- Calibration testing will be minimized after the development contractors acceptance tests are completed and no calibration tests will be permitted after the payload final instrument alignment activities are completed in the KSC-SPF.

- GSE will be designed to withstand the same transportation environments as the FSE or instruments.

- Some GSE will be designed for specific use at Level III, II, and I integration levels and will not be used at the Level IV site.

A ground operations functional flow was prepared which was compatible with the GSE/Facility groundrules. This flow identifies the major functional operations and their location at each integration level (see Section 3.3). Utilizing the Ground Operations Functional Flow a matrix was prepared to identify the tasks necessary to accomplish each operational function. After identification of these tasks, studies were performed to identify what generic type of GSE/Facilities would be required to satisfy the tasks. Tasks were identified for the individual instruments, FSE, the major Spacelab hardware alone, and for the various integrated assemblies. Comparisons of the GSE/Facility capabilities and requirements are summarized in Table 5.1.2-1. The table includes vertical columns identifying the task, the GSE/Facilities required to satisfy the task, the location where the GSE/Facility is needed and the supplier. Suppliers are identified as the development contractor, the GSFC, the prime contractor, MMSE, Spacelab, Orbiter, Launch Site, or the specific facility identified in the location column. Entries into the table are made one time only, even though some items may be required to satisfy more than one task. The remainder of this section will discuss examples of some of the GSE/Facilities which appear in the table in the same order as the table's task sequence.

<u>Transportation</u> - The development contractor will provide, for the purpose of transportation of AMPS instruments and FSE, a shipping container, environmentally controlled as required, and will also arrange shipment to the Level IV integration site. Prior to placement in shipping containers instruments will be placed in plastic bags to maintain cleanliness during shipping and handling operations. Shipping containers for pallets and racks are shown in Figures 5.1.2-2 and 3.

Table 5.1.2-1 AMPS GSE/Facility Task Requirements

Tasks	GSE/Facilities Required	Location: Developer	Level IV	Level III	Level II	Level I	Supplier
Transportation	Shipping Containers/ Plastic Bags						
	Environmental Servicing/Sensing Kit						
	Transporter						
	Instruments, FSE	X	X				Developer
	Pallets, Racks	X	X	X			Spacelab/MMSE
	Spacelab/Pallet				X	X	MMSE
Receiving & Inspection	Facility Airlocks		X	X	X		Facility
	Clean Rooms	X	X	X	X	X	
	General-Purpose Test Equipment (Scopes, Meters, etc.)	X	X	X	X	X	
Inventory/ Storage	Bonded Storage Areas		X				Facility
	Bonded Clean Rooms		X				
Installation/ Handling	Facility Cranes		X	X	X	X	Facility
	Slings						
	Instruments/FSE	X	X				Developer
	Pallets, Racks		X	X			Spacelab
	Strongback (Spacelab/ AMPS)				X	X	MMSE
	Handling Fixtures						
	Instruments, FSE	X	X				Developer
	Pallet Segment Support		X	X			Spacelab
	Rack Handling/Support Kit		X	X			Spacelab
	Spacelab/AMPS (Payload Container)						MMSE
	Pallet Simulator						Prime Contractor (22 1/2-Day Pallet Cycle)
	Instrument Protective Covers		X	X	X		Prime Contractor/ Spacelab
	Mechanical Interface						
	Instrument/FSE Alignment						
	Interface Tooling	X	X				Developer/Prime Contractor

Table 5.1.2-1 AMPS GSE/Facility Task Requirements (Continued)

Tasks	GSE/Facilities Required	Location: Developer	Level IV	Level III	Level II	Level I	Supplier
Installation/ Handling (cont)	Optical Alignment Kit			X			Spacelab
	Optical Cleaning Kit	X	X			X	Developer/Prime Contractor
	Rack & Floor Installation Kit			X	X		Spacelab
	Pallet Mate/Demate Kit			X			Spacelab
Access	Instrument, FSE						
	Pallet Segment Floor Covers		X	X	X		Spacelab/Prime Contractor
	Module Segment Floor Covers				X		Spacelab
	Pallet Workstands		X	X			Prime Contractor (Level IV Modification Items)
	Payload Horizontal Access Kit				X		Spacelab/MMSE
	Instrument Access Kit		X	X	X		Prime Contractor/ Spacelab
	Integration & Check-out Stand			X	X		Spacelab
	Payload Changeout Room		X	X	X		Prime Contractor/ Spacelab
Interface Verification	Facility Power/Services	X	X	X	X	X	Facility
	Power Conditioning Units	X	X				Developer
	GSE Cables	X	X	X	X	X	Facility-Peculiar (e.g., Prime Contractor-Soft Mating Cables, Simulator Cables)
	EMI Diagnostic Equipment		X		X		Prime Contractor/ Spacelab
	Service Kits/Plumbing Lines						
	Freon	X	X	X			Facility
	Gaseous Nitrogen					X	Facility
	Liquid Helium		X	X	X	X	Facility
	Gaseous Neon		X	X		X	Facility
	Leak Check	X	X	X			Facility/MMSE

Table 5.1.2-1 AMPS GSE/Facility Task Requirements (Concluded)

Tasks	GSE/Facilities Required	Location: Developer	Level IV	Level III	Level II	Level I	Supplier
Interface Verif. (cont)	Magnetic Field Generator		X		X		Prime Contractor
	Pyrotechnic Initiator Test Kit		X				Prime Contractor
	Computer						
	IBM 370 or Equivalent & Ancillary Equipment		X	X	X		GSFC (GFE)
	GSE Software	X	X		X		Developer/Prime Contractor/Spacelab
Integration Tests	Vibro/Acoustics						
	Instrument Systems		X				Prime Contractor
	Pallet Level		X				Prime Contractor/GSFC
	Thermal Vacuum						
	Instrument Systems	X	X				Developer/GSFC/Prime Contractor
	Instrument/FSE Calibration	X	X	X			Developer
	Instrument/FSE Data Readout	X	X	X			Developer
	Simulators						
	CSS or Equivalent		X				Spacelab
	OIA			x	x		Spacelab
	Spacelab ATE			X	X		Spacelab
	LPS				X	X	Launch Site

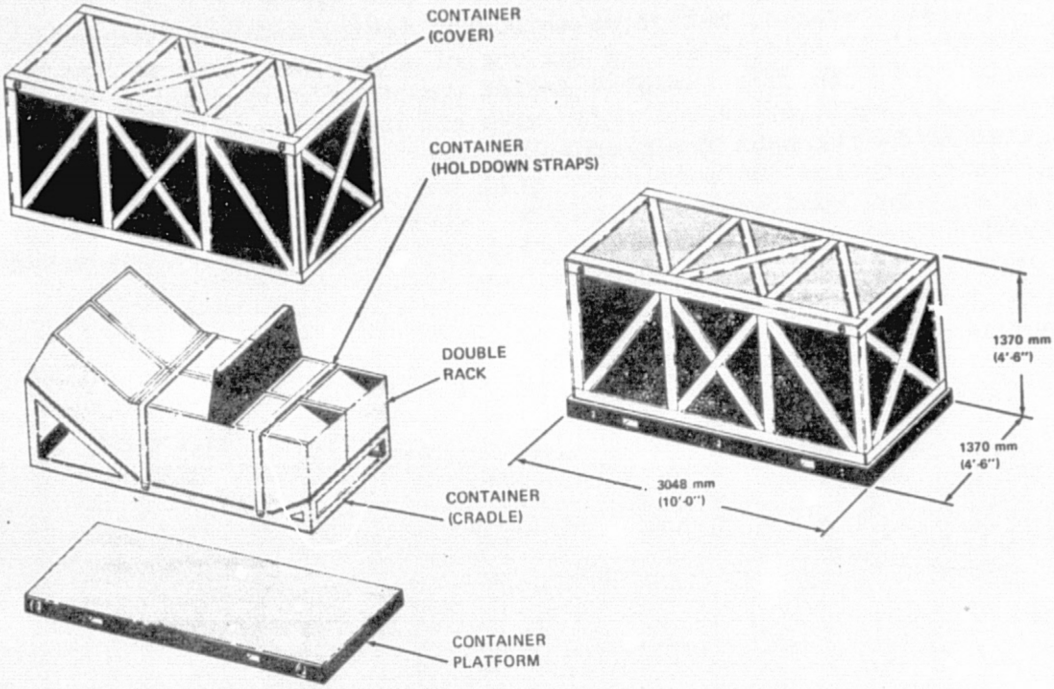


Figure 5.1.2-2 Rack Shipping Container

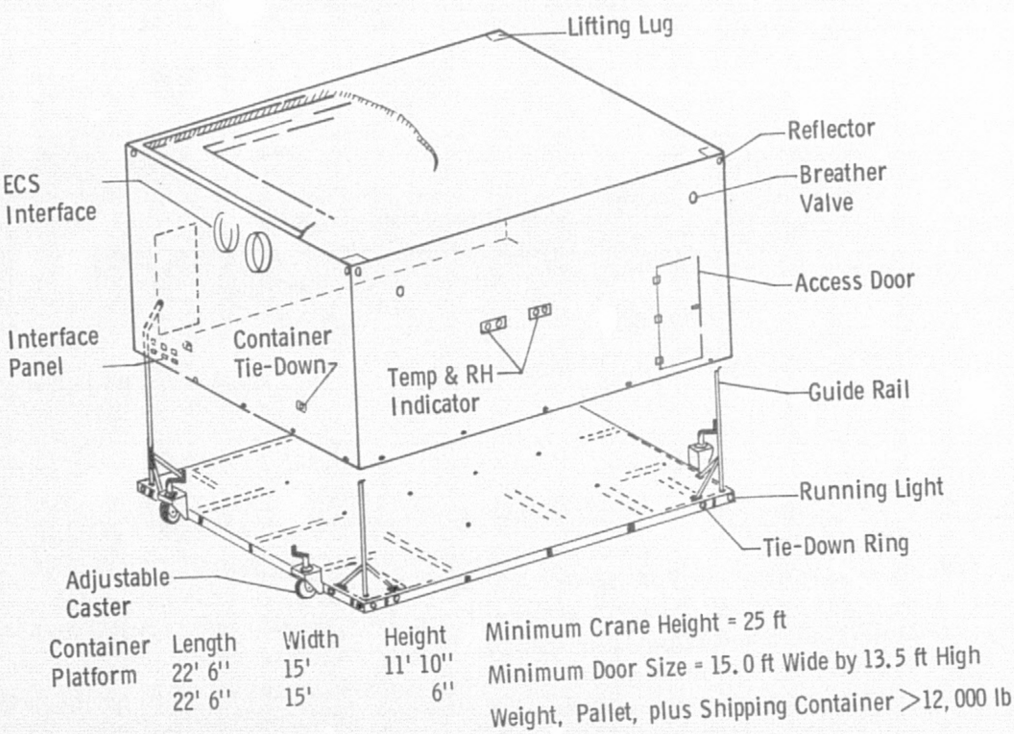


Figure 5.1.2-3 Hard Container - Large System

The transportation method selected for moving instrumented pallets from the Level IV site to KSC is the use of a low-boy truck as shown in Figure 5.1.2-4. The instrumented pallet was of primary concern because it has been identified as the major size and weight item. This approach was selected by the NASA Transportation Committee, a group with multi-center participation who considered road, rail and air as candidate approaches. The road transportation was chosen on the basis of lowest cost without compromise to other operational parameters, such as schedule, security and number of handling operations.

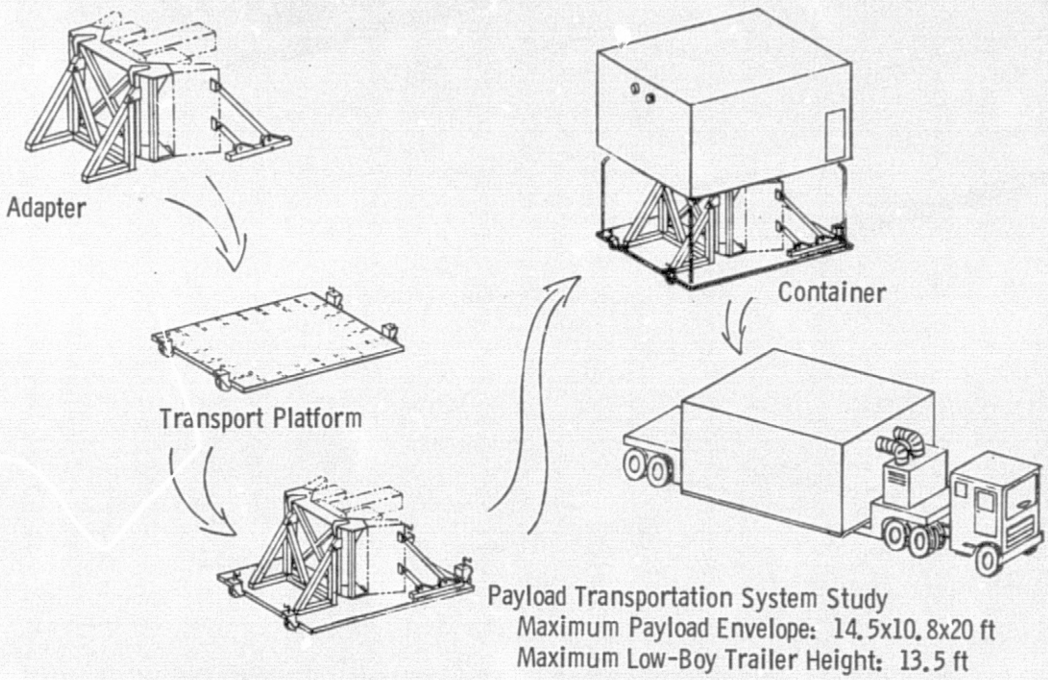


Figure 5.1.2-4 Road Transporter for Instrumented Pallets

Receiving and Inspection - In order to support receiving, inspection and subsequent test activities clean rooms are required at each site. Clean rooms required for Level IV and subsequent testing must be able to handle large volumes, so they must be equipped with airlocks. The study included an investigation of facility requirements based upon the largest volume, envelope weight, and cleanliness level that must be provided. Of particular interest were the prime contractor's Level IV site and the KSC site for supporting off-line activities. Based upon

the instruments, FSE, and integrated AMPS requirements we investigated facilities provided by the instrument developer, the GSFC, the prime contractor and the KSC.

Installation, Handling and Access - In order to satisfy the Level IV activities we identified an overall floor space allocation of 30' x 50' for 3 pallets and 30' x 70' for 5 pallets, as shown in Figure 5.1.2-5. Door size, crane height, crane capacity, ceiling height requirements were established by Figure 5.1.2-3; cleanliness requirements to date have not been established beyond class 100,000, although cleaner environments are anticipated for instrument/system assembly facilities. An investigation of facilities at the prime contractor's site showed that the Space Support Building's High Bay Area shown in Figure 5.1.2-6 could accommodate the major Level IV activities and other laboratories could support additional Level IV activities, such as the Acoustic/Vibration Laboratory, the Space Simulation Laboratory, and the Man-Computer Interaction Laboratory.

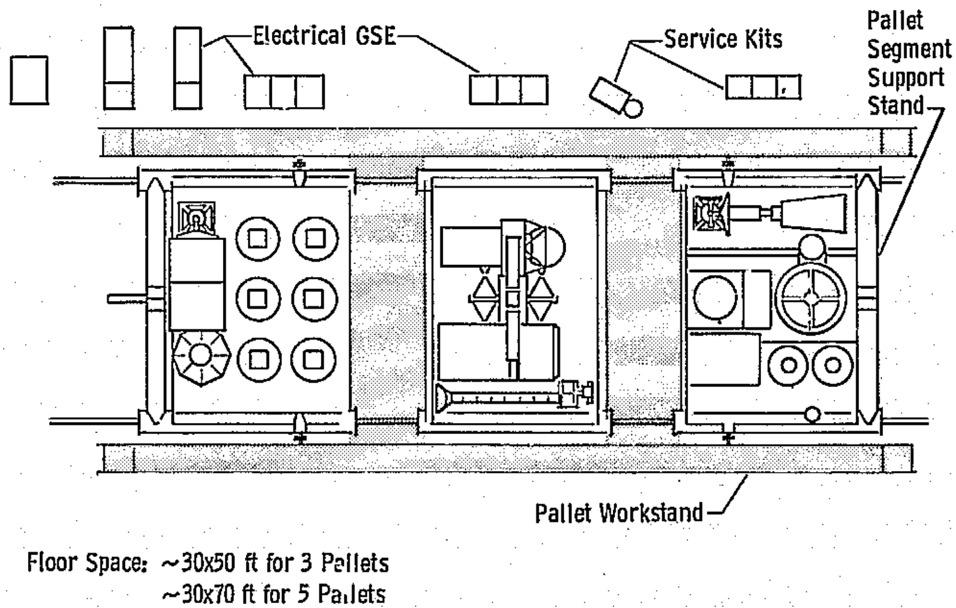


Figure 5.1.2-5 Level IV Floor Space Allocation

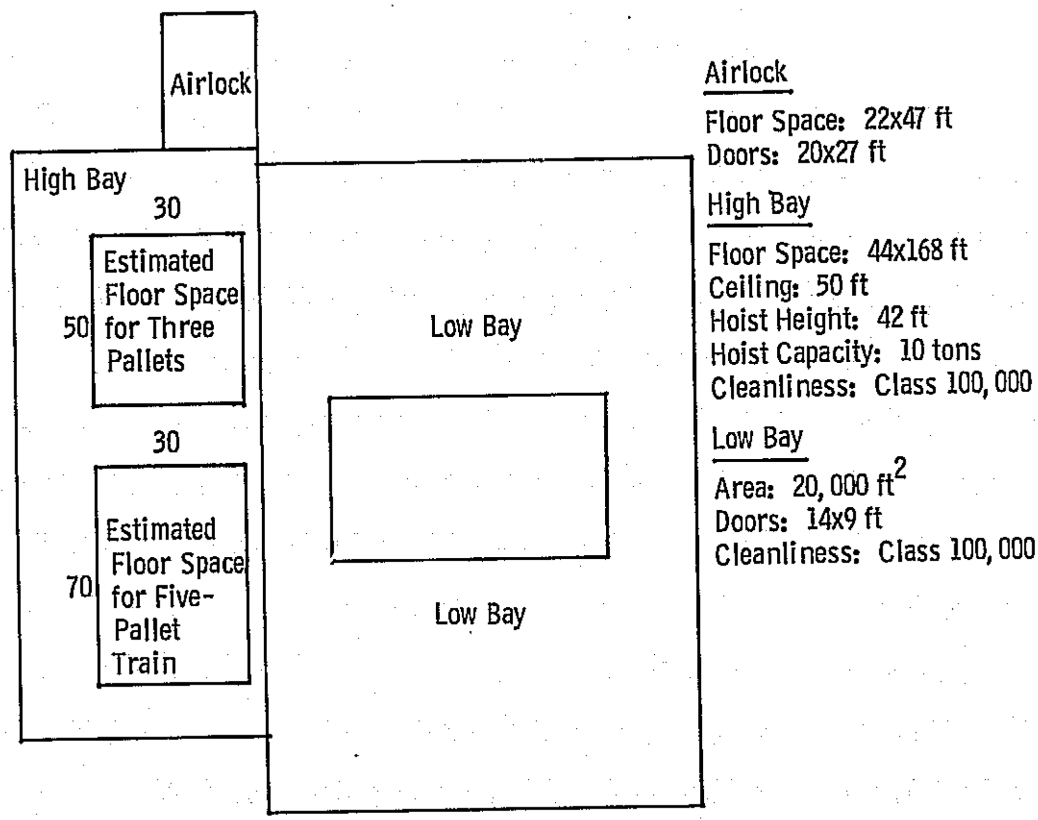
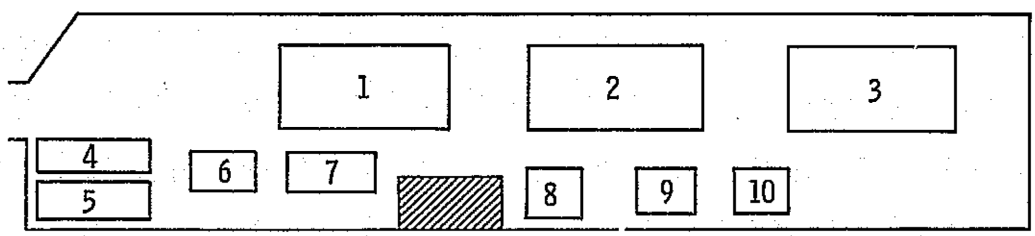


Figure 5.1.2-6 Martin Marietta Facility Capabilities - Space Support Building

The KSC facilities and a summary of their functions required to support AMPS activities are shown in Table 5.1.2-2. An investigation of the O&C Building's Layout, presented in Figure 5.1.2-7, showed that limited off-line activities could be supported, especially for earlier AMPS flights. However, when the Space Shuttle is scheduled to fly ten-to-twelve times annually the O&C location will not be adequate for support of AMPS activities. (Note: AMPS off-line Level IV activities would have to be supported by the areas designated as Items 5 and 6 in Figure 5.1.2-7.

Table 5.1.2-2 KSC Facilities Required for AMPS Support

Facility	Function
Orbiter Processing Facility (OPF)	Safing Spacelab Subsystems Install/Remove Payload from Orbiter Level I Integration and Checkout
Vehicle Assembly Building (VAB)	Mate Orbiter with External Tank and Solid Rocket Boosters
Payload Changeout Room (PCR)	Vertical Assembly of Payload with Orbiter Working Facility for Payload Pad Operations
Operations and Checkout (O&C)	Mate/Demate Spacelab with Payload Perform Compatibility Testing (Level II, Level III) Limited Off-Line Activities
Hangars, AE, AM, and AO SAEF 1 and 2	Off-Line Activities (PHF) Receiving/Inspection, High-Bay Clean Rooms Specific Payload Activities - Subsystem Tests



Items	Description	Area, ft
1, 2, 3	Spacelab Test Stands	30x70
4	Tunnel Area	20x40
5	Three-Pallet Stand	20x40
6	Two-Pallet Stand	20x26
7	Rack Stand	20x26
8	Igloo Area	10x10
9, 10	End Cone Stands	16x10

Floor Space: ~ 22,000 ft^2

Ceiling: 80 ft

Hoist Height: 50 ft
Hoist Capacity: 27 1/2 tons

Cleanliness: Class 100,000

Doors: 40x80 ft

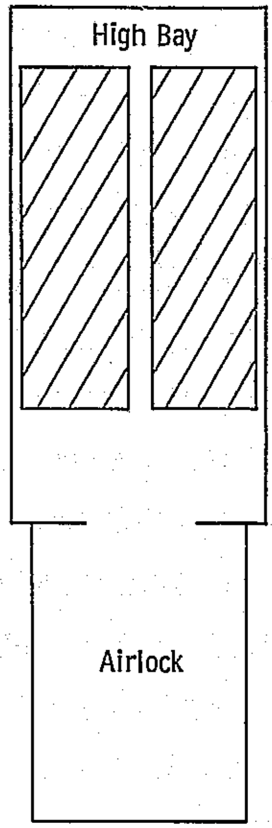
Figure 5.1.2-7 KSC Facilities - O&C High-Bay Layout

Additional KSC facilities were investigated to support the AMPS program off-line activities and they are shown in Table 5.1.2-3 as potential Payload Handling Facilities. Because of scheduled activities such as, the free-flyer activity in Building AO and the IUS activity in Hangar SAEF-1; limited floor space in Building AE; limited door space, no airlock, and insufficient cleanliness in Building AM; too low a ceiling height and limited crane capacity in Hangar S; Hangar SAEF-2 shown in Figure 5.1.2-8, was chosen as the prime candidate for the KSC Payload Handling Facility. Building AO shown in Figure 5.1.2-9 was recommended as a backup since its present scheduled use appears limited.

Table 5.1.2-3 Potential Payload Handling Facilities

Building	Cleanliness Class	Floor Space, ft	Ceiling Height, ft	Crane Height, ft	Crane Capacity, ton	Door Size, W x H, ft
AE	10,000	43x49	34	33.7	10	16x36.5
AM, North South	Industrial Class	35x70 48x50	42 42	35.0 35.0	5 5	15x34 15x20
AO	100,000	46x175	46	45.0	2 at 10	25x40
Hangar S, North South	100,000 100,000	30x43 23x56	22 24	20.0 20.0	2 5	16x20 16x20
Hangar SAEF 1 SAEF 2	100,000 100,000	120x150 99x49	105 74	96.0 67.0	25 10	40x86 21x38

The pallet segment support stand and the pallet work stand (identified in Table 5.1.2-1) provide installation and access capability for the Level IV activities as shown earlier in Figure 5.1.2-5. Additional items identified in the table to support installation, access, and interface verification are shown in Figures 5.1.2-10 through 14 and will be supplied by either MMSE, Spacelab or the launch site.



High Bay

49x99-ft Floor Space
21-ft Wide by 38-ft High Door

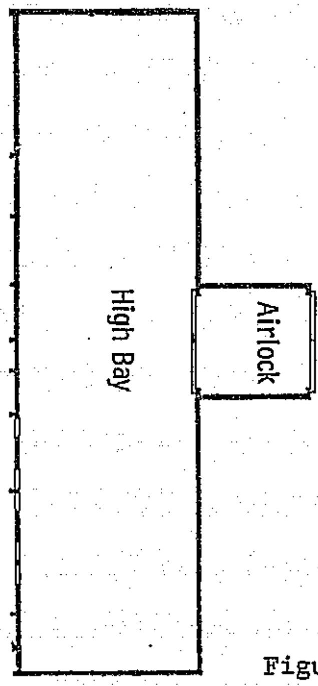
Airlock

41x58-ft Floor Space

Comments

Large Enough for Two Five-Pallet Trains
If Stands (20x66 ft) Partially Disassembled

Figure 5.1.2-8 KSC Facilities - Building SAEF No. 2 (Prime PHF)



High Bay

56x175-ft Floor Space

Airlock

26x29-ft Space
48-ft Ceiling
25-ft Wide by 40-ft High Doors

Comments

Airlock Large Enough for Two-Pallet Train
Planned for Use by Free-Flyers

Figure 5.1.2-9 KSC Facilities - Building AO (Backup PHF)

Figure 5.1.2-10 illustrates typical handling slings, the strongback required for handling the entire AMPS/Spacelab payload and the Payload Handling container. Figure 5.1.2-11 illustrates access equipment including equipment for access when AMPS is in the canister for shipment to the Level I site, the Payload Assembly/Test Horizontal Workstand for Level III/II tests and the Payload Container Access Equipment and the Payload Changeout Room for vertical access while AMPS is on the launch pad. Use of the Payload Changeout Room is required for servicing cryogens for the limb scanning instruments.

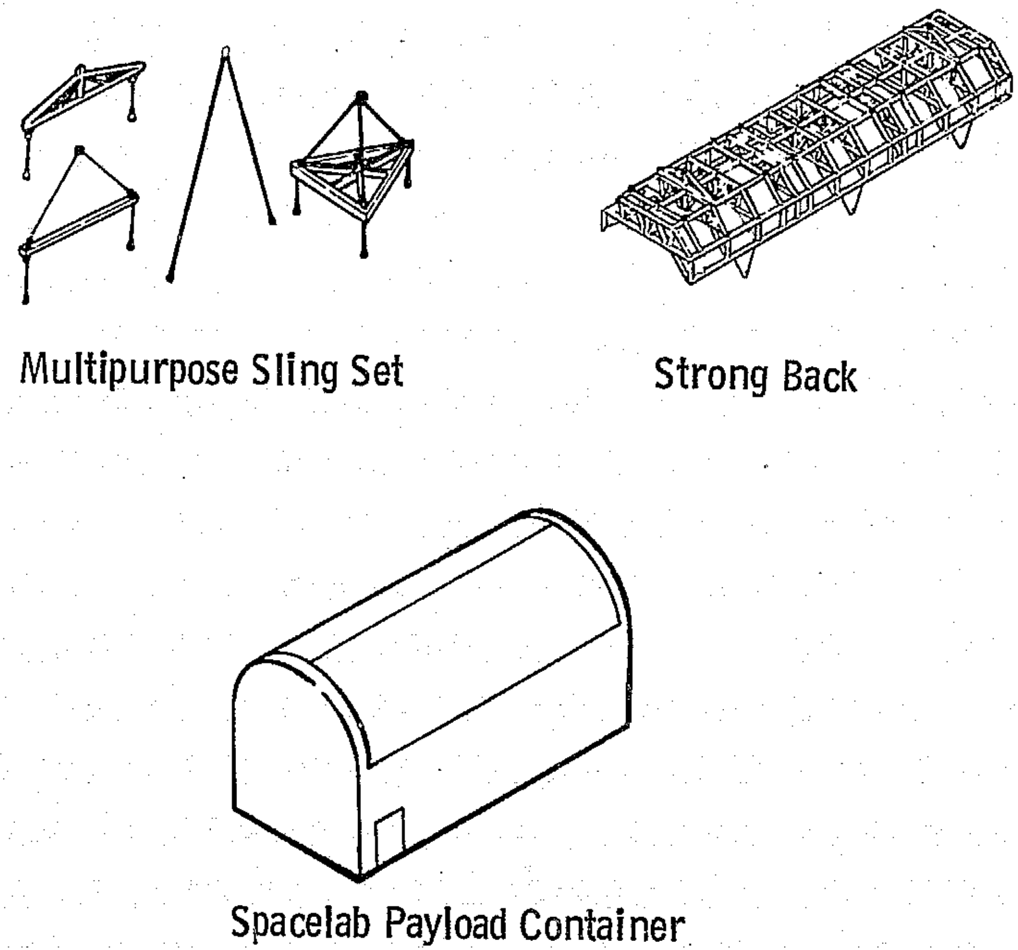
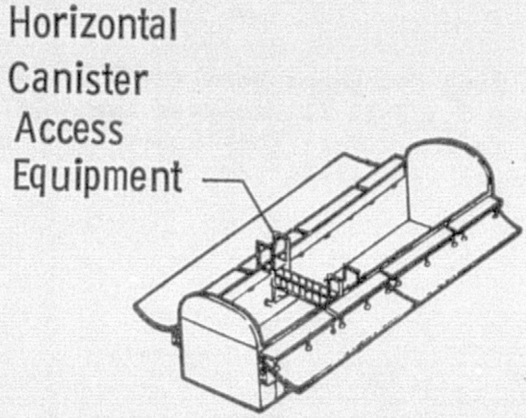
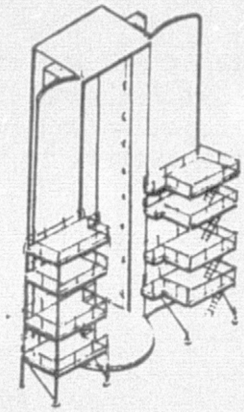


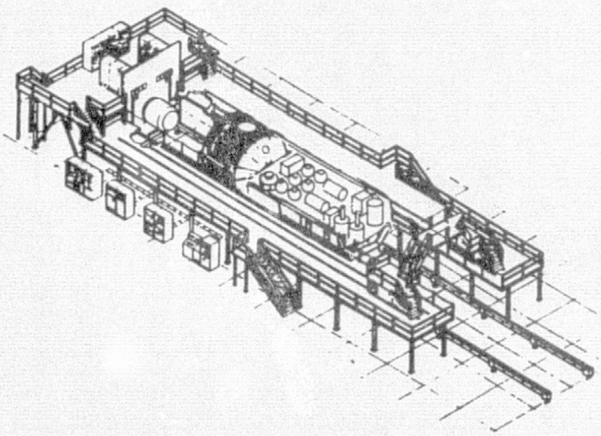
Figure 5.1.2-10 Installation Handling MMSE



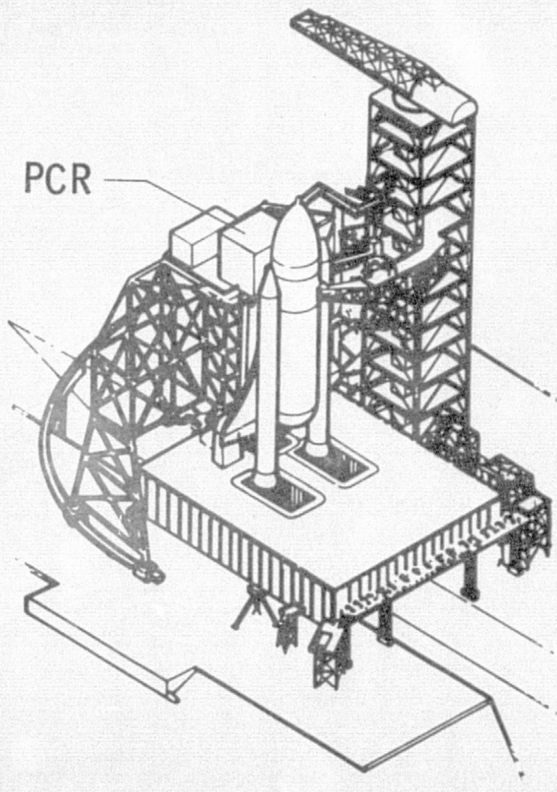
Payload Horizontal Canister Access Equipment



Payload Container Access Equipment



Payload Assembly/Test Horizontal Work Stand



Payload Changeout Room

Figure 5.1.2-11 Access Equipment Required by AMPS

Details of the use of the Payload Changeout Room (PCR) are shown in Figures 5.1.2-12 and 13. Figure 5.1.2-12 illustrates the AMPS/Orbiter attached to the PCR prior to the PCR airlock doors being opened and while the airlock is being purged with clean air.

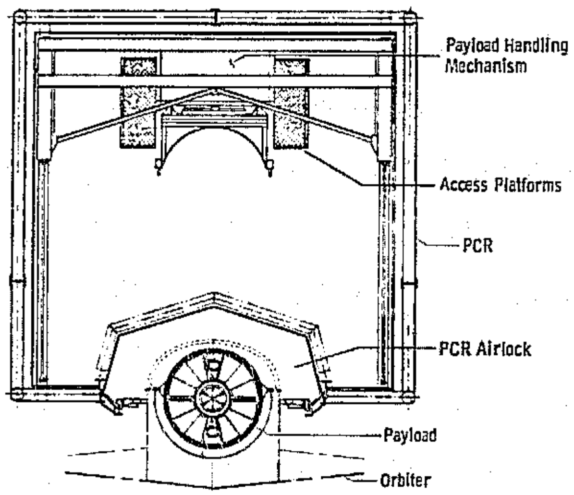


Figure 5.1.2-12 KSC Facilities - AMPS/Orbiter Attached to PCR

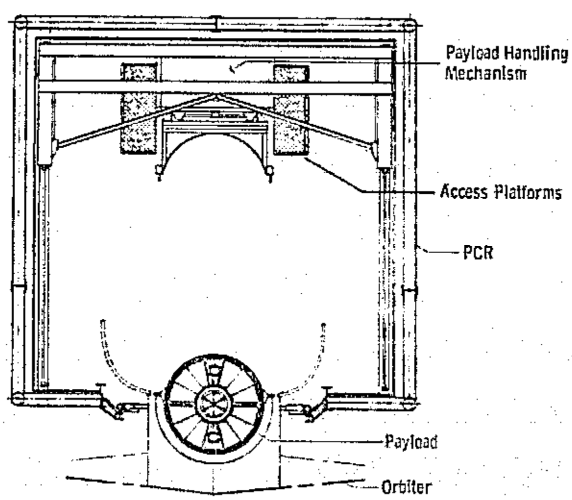


Figure 5.1.2-13 KSC Facilities - PCR Access to AMPS

Figure 5.1.2-13 shows the Payload doors opened and the Access Platforms prior to their being rolled into place at the payload for cryogen servicing. (The Airlock details have been omitted for simplification of the figure). This operation occurs in the final hours prior to lift-off, so the ground cryogen lines can be serviced, topped off, and disconnected as late as possible prior to flight. Figure 5.1.2-14 illustrates service kits required for installation and verification of instruments and FSE.

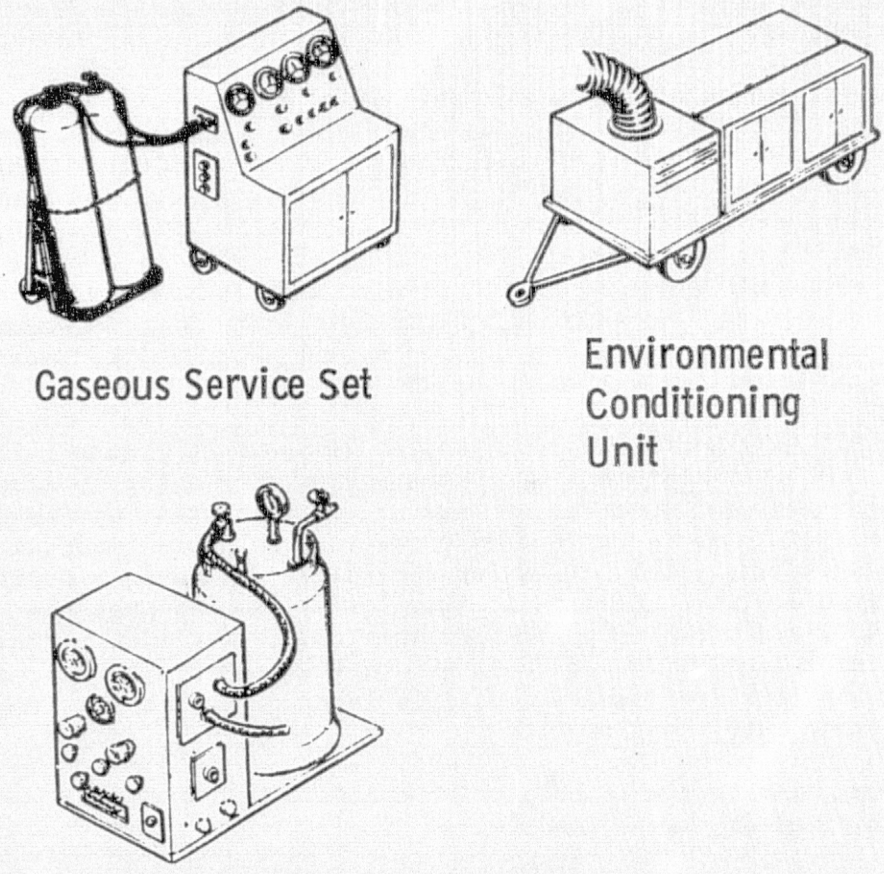


Figure 5.1.2-14 Verification Service Kits

<u>Verification</u> - For support of integrated tests after entering the in line integration flow the Core Segment Simulator (CSS), the Orbiter Interface Adapter (OIA), the Spacelab Automatic Test Equipment (ATE), and Launch Processing System (LPS) have been identified in Table 5.1.2-1. A description of these items along with functional block diagrams is contained in the Spacelab Accommodations Handbook and the Launch Site Accommodations Handbook. The CSS will be used during Level III activities prior to assembly of the AMPS Payload to the Spacelab Core Segment. The OIA will simulate the Orbiter to Spacelab electrical interfaces and will link directly to Spacelab during Level II testing. The Spacelab ATE will interface with the Spacelab or OIA and perform test sequences and data acquisition, recording, decommutation, evaluation, display, printout and an operator's interface verification console. The LPS will provide test monitoring of payload functions during the Shuttle on-line activities.

Because of the potential unavailability of the Spacelab provided GSE for pre-Level III verification and the need for specialized FSE checkout equipment, provision of electronic GSE is envisioned. The electrical GSE identified is that required to support Level IV testing at the contractor's facility as well as at the KSC payload handling facility. Two types of tests are planned: 1) verification of the instrument to Spacelab interface and 2) integration and checkout of the complete AMPS payload (soft mated). To facilitate Item 1) it is assumed the instrument developer will provide his unique GSE developed during the instrument design, development and checkout phase.

An overall diagram of the EGSE concept is shown on Figure 5.1.2-15. The system is an automatic test set using a ground computer to simulate the Spacelab experiment computer operation. A CRT display and keyboard are also provided to simulate onboard operations but more importantly to provide operational flexibility for special tests and test modifications. To provide the necessary fidelity, it is assumed that the experiment RAUs are provided by the program to verify remote instrument control and data acquisition and monitor. All equipment is designed for mobility to facilitate use at multiple locations. The required EGSE are categorized into two groups:

1) Equipment to simulate Spacelab module electronics and services provided to the payload:

    Computer
    I/O and interface electronics
    Digital multiplex simulator
    Measurement and command interface panel
    CRT & keyboard
    Timing subsystem
    Power supply and distributor

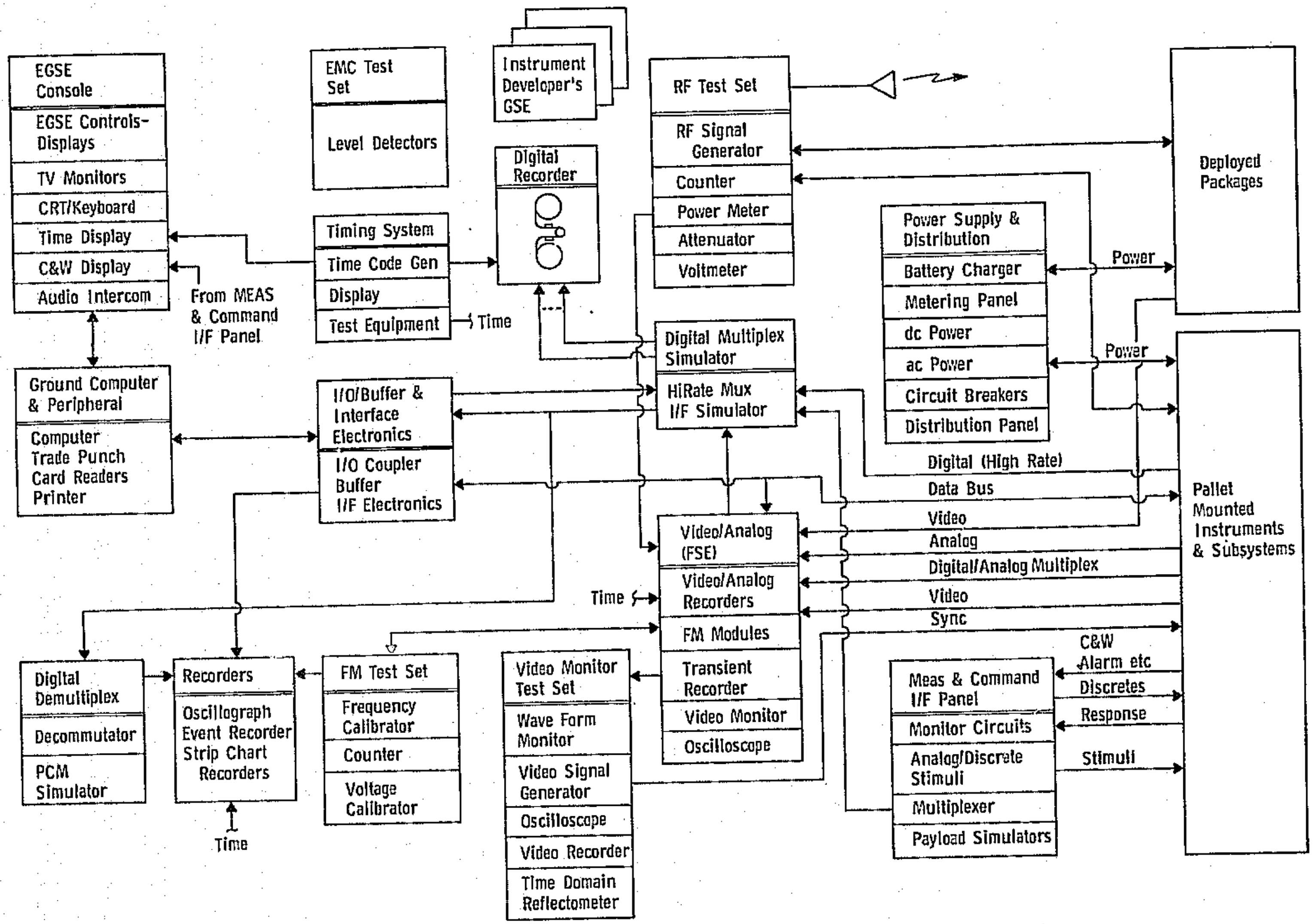


Figure 5.1.2-15 Electrical Ground Support Equipment (EGSE)

2) Test equipment to calibrate and verify flight hardware operations:

Video monitor test set
FM test set
RF test set
Recorders
Digital demultiplexer
EMC test set

Direct EGSE to instrument interfaces are provided primarily in the area of power distribution, hardwire measurement and control interfaces, and science data that would normally interface with the high rate multiplexer. All such GSE interface integrity would be verified before actual hookup. In addition, interfacing circuitry will be designed to protect the flight instruments from EGSE failures. Where flight support equipment or any subsystem cannot be operated because of one G limitation, constraints on equipment deployment, pyro initiation, etc, equivalent simulators and circuitry will be provided to verify its operation.

Primary control is provided at the EGSE console whereby computer control or manual test operations are provided. All caution, warning signals are displayed at this console which together with the CRT displays ensures safe operation of the payload. The computer/software will execute the test sequences at the discretion of the console operator. The software model will include automatic sequencing, data processing, limit checks and automatic shutdown for off-nominal operation of critical functions.

The experiment data bus, RAU and the computer are used to operate, monitor and acquire low rate data. An I/O/buffer is provided to properly interface the computer with the RAUs. High rate data and other digital data required to support Level IV testing are routed to the digital multiplex simulator. This unit will provide the same interface circuitry as incorporated in the high rate multiplexer but will not interleave multiple inputs.

Video/analog data will be routed to the video analog rack which will be that used in flight. This rack is discussed in greater detail in Section 4.5. To support checkout of the video system a video monitor test set is provided. This unit provides the necessary test signals and monitor equipment required to support the acceptance test. The FM test set is similarly required to generate calibration signals and monitor the module outputs. A digital demultiplexer is provided primarily to decommutate and verify operation of the PCM system on the deployed packages. A permanent record of all data are provided by the various recorders which are interconnected to the computer system, FM and digital test sets.

Since all diagnostic packages and the pallets will have RF transmitters and receivers, an RF test set is required to verify transmitter characteristics; ie, RF frequency, power and receiver sensitivities. This unit will be capable of open loop and closed loop operations. The measurement and command panel interface is designed to provide unique stimulus and response circuitry required to verify the electrical/electronic system as required by the Level IV test plan. In addition, it will simulate functions that cannot be operated, such as equipment ejection, platform operation, etc. These signals will be multiplexed for recording and limit checks by the computer. The EMC test set will monitor critical circuits to ensure that interference levels are below an established threshold. Circuit detectors will be interconnected to the test set where adjustable trip level circuitry will be provided.

5.1.3 Maintenance and Refurbishment

The maintenance approach selected for the AMPS payload requires that all planned maintenance be performed on the ground. Performance of inflight maintenance, except on a limited contingency basis, is not cost effective. This is primarily because of the short mission duration, the added crew training requirements, and the added costs of spares and testing. Performance of inflight maintenance on selected critical items located in the pressurized area should be considered for mission and safety critical failures.

The maintenance and refurbishment activities will be accomplished at the KSC/Payload Handling Facility except for major modifications to the instruments, FSE, or GSE which will be accomplished at the facility of the hardware developer. The maintenance and refurbishment activities that will be accomplished at the Payload Handling Facility include: decontamination of the payload; performance tests to determine the status of payload elements required for the next flight; demating of the pallet train; removal of instruments, FSE, payload specialist station, and racks; minor modification and refurbishment of payload elements required for the next flight; installation of new payload elements for the next flight; and verification testing of the refurbished payload. Analysis will be continued during the detail design of the AMPS payload to insure selection of the optimum maintenance approach and to establish detailed maintenance and refurbishment support requirements.

5.1.4 Logistics

Preliminary logistics support analyses have been conducted to determine the location of the AMPS refurbishment site, facilities required to support maintenance and refurbishment, and preferred methods of transportation for the AMPS payload elements between sites.

Preliminary analysis indicates that KSC would be the most attractive refurbishment site for AMPS/Labcraft payloads. The concept of multi-flights each year dictates a short turn-around time which requires that much of the refurbishment activity be accomplished near the launch site and without removal from the integrated pallet if possible. This includes all maintenance and refurbishment activities except for major modifications which will be accomplished at the hardware developer's facility. Also, based on the preliminary analysis, it is recommended that a dedicated AMPS/Labcraft Payload Handling Facility be provided. This facility should include provisions for storage of spares and new instruments for subsequent flights.

A preliminary analysis was conducted to determine the method of transporting the AMPS payload elements from the prime contractor facility to KSC. It was concluded that the payload should be shipped as individual pallets and that this could be accomplished satisfactorily either by air or by ground transportation. This analysis will be

continued during the next program phase and the most cost effective method selected.

In addition to the transportation study, continuing logistics support analyses will be conducted to establish detailed requirements for consumables, tools, and spares; handling, storage, and packaging; and maintenance/operating personnel training.

5.1.5 Reliability

The basic reliability requirements for the AMPS payload hardware is that no single failure point shall exist that can result in loss of life or serious injury to personnel, mission termination, or major degradation causing loss of more than 50% of the instrument/experiment data. Since there were no specified numerical reliability or operating lifetime requirements, it was concluded that redundancy and/or safety margins would be required only where necessary to satisfy the above single failure point (SFP) criteria.

Using the established SFP criteria the AMPS payload configurations for Flights 1 and 2 were evaluated and the potentially critical hardware elements identified. Table 5.1.5-1 is a preliminary list of the AMPS payload critical items. The pyrotechnics, high voltage power supply, laser sounder, and electron beam accelerator will require some form of redundancy or other built-in protection against inadvertant operation. For the pressure vessels, structural safety margins will be required. The extended instruments will be required to have redundant retraction mechanisms or a method of jettisoning the instrument to allow closure of the payload bay doors. For the electrical distributors, SIPS, RF terminal, and high rate multiplexer, where failure can result in loss of a significant amount of experiment/instrument data, some form of redundancy or work around capability will be required. These requirements/criteria will be updated and specified in detail for each item at the beginning of AMPS detail design based on the results of the failure mode and effects analyses (FMEA) that will be performed. In addition, operating life requirements will be developed for all hardware reused for multiple flights. This will be determined from life cycle cost analyses when the hardware is better defined in detail design.

Table 5.1.5-1 AMPS Payload Reliability Critical Items

CRITICAL ITEM	SUBSYSTEM	BASIS FOR CRITICALITY
High Rate Multiplexer	FSE/DMS	Significant Experiment/ Instrument Data Loss
Electrical Power Distributors	FSE/EPS	Significant Experiment/ Instrument Data Loss
Pyro Initiators and Controls	FSE/EPS	Personnel Safety
High Voltage Power Supply	FSE/EPS	Personnel Safety
RF Terminal	FSE/COMM	Significant Experiment/ Instrument Data Loss
SIPS	FSE/ACPS	Significant Experiment/ Instrument Data Loss
Retract Mechanisms For Instruments II-3, II-7, II-9, II-10, II-12	FSE/S&M	Personnel/Vehicle Safety
Pressure Vessels In Instruments I-21, II-1, II-7, II-10, III-3	Instruments	Personnel/Equipment Safety
Laser Sounder	Instruments	Personnel Safety
Electron Beam Accelerator	Instruments	Personnel Safety

5.1.6 Safety

Preliminary safety analyses were performed on the AMPS payloads for Flights 1 and 2 to identify and resolve potential payload and interface safety hazards as early as possible in the program. These analyses were performed on the instruments, FSE, GSE, and interfaces, and included identification of potential hazards during all ground and flight operations; definition of the energy source and release mechanisms associated with these hazards; determination of the possible effect of the hazards on both personnel and equipment; and definition of proposed methods of elimination and/or controlling these hazards. During the subsequent design and development phases of the program, detailed hazards analyses based on failure mode and effects analyses (FMEA) and safety checklists will provide specific safety design requirements as well as procedural requirements for testing and operations safety.

The potential hazards identified for Flights 1 and 2 payloads are summarized in Tables 5.1.6-1 and -2 which also include the potential effects, energy source, and release mechanisms for each of the hazards.

Table 5.1.6-1 AMPS Flight 1 Potential Hazards Summary

Hazard	Energy Source	Release Mechanism	Operation/Phase	Effect
Electrical Energy - High-Voltage/Current	I-1 Lidar I-9 Electron Beam FSE High-Voltage Power Supply	Personnel Contact With High Voltage	Ground Checkout and Prelaunch Operations Maintenance and Refurbishment Operations	Possible Injury to Personnel - Shock, Burns
Pressure Energy - High-Pressure Gas	I-21 Chemical Release System (SS 1) II-1 Cryocooled Limb Scanner II-10 Far-IR Spectrometer III-3 Beam Diagnostics (Level I)	Pressure Vessel Ruptures Due to Increase in Temperature	Ground Checkout and Prelaunch Operations Maintenance and Refurbishment Operations Flight Operations	Possible Injury to Personnel Possible Damage to Equipment
Low Temperature - Supercold Liquids	II-1 Cryocooled Limb Scanner II-7 Cryocooled Limb Scanner II-10 Crocooled Interferometer Spectrometer	Cryogenic Container and/or Lines Leak	Ground Checkout and Prelaunch Operations Maintenance and Refurbishment Operations Flight Operations	Possible Injury to Personnel Possible Damage to Equipment
High-Power Laser Beam	I-1 Lidar	Personnel Contact With Laser Beam and/or Reflected Laser Beam	Ground Checkout and Prelaunch Operations Maintenance and Refurbishment Operations	Possible Injury to Personnel - Burns, Eye Damage
Pyrotechnic Devices - Inadvertent Operation of Device Used to Release Gas	I-21 Chemical Release System (SS 1)	Inadvertent Application of Power to Device	Ground Checkout and Prelaunch Operations Maintenance and Refurbishment Operations Flight Operations	Possible Injury to Personnel Possible Damage to Equipment
Instrument Ejection/ Separations Mechanisms Inadvertent Operation	I-21 Chemical Release System (SS 1) III-25 EMI Package FSE	Inadvertent Application of Power	Ground Checkout and Prelaunch Operations Maintenance and Refurbishment Operations Flight Operations	Possible Injury to Personnel Possible Damage to Equipment
Instrument Deploy/ Retract Mechanisms Failure to Retract	II-3 OBIPS II-7 Limb Scanner II-9 Near-IR Spectrometer II-10 Interferometer Spectrometer	Deploy/Retract Mechanism Breaks or Jams with Instrument Deployed	Flight Operations	Inability to Close Payload Bay Doors and Possible Loss of Orbiter Entry Capability
Chemical Release Module or Fragments of Chemical Release Module Impacts Shuttle	I-21 Chemical Release System (SS 1)	Module Is Ejected from Payload Bay ANL Is Opened to Release Gas Using Pyro Device	Flight Operations	Possible Damage to Space Shuttle and/or Payload Possible Crew Injury

Table 5.1.6-2 AMPS Flight 2 Potential Hazards Summary

Hazard	Energy Source	Release Mechanism	Operation/Phase	Effect
Electrical Energy - High-Voltage/Current	I-1 Lidar FSE	Personnel Contact with High Voltage	Ground Checkout and Prelaunch Operations Maintenance and Re-	Possible Injury to Personnel - Shock, Burns
Pressure Energy - High-Pressure Gas	II-7 Cryocooled Limb Scanner II-10 Far-IR Spectrometer	Pressure Vessel Rupture Due to Increase in Temperature	Ground Checkout and Prelaunch Operations Maintenance and Re-furbishment Operations Flight Operations	Possible Injury to Personnel Possible Damage to Equipment
Low Temperature - Supercold Liquids	II-7 Cryocooled Limb Scanner II-10 Cryocooled Inter-ferometer Spectrometer	Cryo Container and/or Lines Leak	Ground Checkout and Prelaunch Operations Maintenance and Re-furbishment Operations Flight Operations	Possible Injury to Personnel Possible Damage to Equipment
High-Power Laser Beam	I-1 Lidar	Personnel Contact with Laser Beam and/or Reflected Laser Beam	Ground Checkout and Prelaunch Operations Maintenance and Re-furbishment Operations	Possible Injury to Personnel - Burns, Eye Damage
High Temperature - Superhot Gas (2200° to 3200°C) Toxicity - Ba and BaO	I-21 Chemical Release System (SS 2)	Inadvertent Ignition of One or More Barium Canisters	Ground Checkout and Prelaunch Operations Maintenance and Re-furbishment Operations Flight Operations	Possible Injury to Personnel - Burns, Toxic Possible Damage to Equipment Possible Damage to Shuttle
Instrument Ejection/ Separation Mechanisms Inadvertent Opera-tions	I-21 Chemical Release System (SS 2) II-12 SS 3 100-m Dipole Antenna III-17 Deployable Test Body	Inadvertent Application of Power	Ground Checkout and Prelaunch Operations Maintenance and Refurb-ishment Operations Flight Operations	Possible Injury to Personnel Possible Injury to Equipment
Instrument Deploy/ Retract Mechanisms Failure to Retract	II-3 OBIPS II-7 Limb Scanner II-9 Near-IR Spectrometer II-10 Interferometer Spec-trometer I-12 RF Dipole Antenna	Deploy/Retract Mech-anism Breaks or Jams with Instrument De-ployed	Flight Operations	Inability to Close Orbiter Payload Bay Doors and Possible Loss of Orbiter Entry Capability

The most critical of the potential hazards are those resulting from the pressure vessels, the pyrotechnic devices, and the mechanisms to retract extended instruments. These hazardous conditions must be eliminated or controlled by the design. The hazards to ground personnel during checkout, prelaunch, maintenance, and refurbishment operations must be eliminated by design or controlled by procedures.

A detailed hazards analysis will be performed on the instruments, FSE, GSE, and interface during detail design of the AMPS payload. All hazards that are not eliminated will be tracked on a continuing basis throughout the remainder of the program.

5.1.7 Verification

This section presents results from the analysis performed to define a verification approach for the AMPS program. Discussion includes key factors and considerations which apply to STS/Spacelab payloads and how they influence the approach. The recommended approach is presented starting with an overview and followed by a more detail discussion of its elements.

5.1.7.1 General Discussion

The STS era introduces new features and modes of operation for space payloads. Some of these affect, or have the potential of affecting, the amount of testing required for adequate flight assurance of the payloads. In most cases, however, a closer look indicates the need for intricate tradeoffs by the user involving data which are not readily available. Reusability is such a feature. It implies a "second chance" through reflight in cases of failure, but the cost of such reflight may exceed that of initial savings from reduced testing. It also complicates the establishing of verification requirements for cases of multiple flight equipment.

Another feature available for exploitation by the user to a greater extent than before is crew participation (mission and payload specialists). It can be planned or used on a contingency basis to correct malfunctions or alter circumstances which may cause malfunctions. Several examples can be cited, one of which is a possible reduction in thermal testing. If, due to unforeseen circumstances, thermal limits of some equipment are approached, the flight plan may be adjusted with minor overall impact.

The STS payload capability allows structural design to higher safety factors and, consequently, reduction in structural test requirements. Although some designs will still necessitate structural testing, substantial savings can be realized here.

Several features will increase the amount of verification required for the STS payloads. The manned flight nature of the carrier imposes stringent safety requirements and attendant verification requirements. The STS/Spacelab/payload combination has new and more interfaces as compared to a single spacecraft. Here again, more layers of integration and checkout increases the amount of verification required.

In summary of the foregoing discussion it can be said that STS features and mode of operation offer some reduction in verification requirements, i.e., thermal and structural testing. More widespread potential savings involve difficult tradeoffs which may be realizable when actual data and experience is gained in the STS operations. Increase in verification requirements will result from manned flight safety requirements and increased amounts of integration involved.

5.1.7.2 Guidelines and Criteria

Past and present space programs have evolved general requirements which state general guidelines and criteria for the identification of verification programs. They are applicable, at most levels of program management and hardware and software assembly. These requirements, as modified for the AMPS program, are stated below:

- o The objective of the verification program is to demonstrate and document that the flight and ground systems satisfy their specification requirements.

- o The AMPS test program shall be an integrated test program. The test management shall ensure this through the continuity in test activities throughout the buildup of system elements. Inherent in planning of the buildup process shall be the objective of:

  - Minimizing test duplication;
  - Maximization of standard tests;
  - Combination of tests;
  - Commonality in utilization of resources;
  - Testing at highest assembly levels practical;
  - Uniformity in handling of information (management, technical).

- o Test emphasis (use of actual test methods) shall be applied towards cost effectiveness through the application of cost/value criteria to system elements in relation to their contribution to mission safety and/or objectives.

- o Analytical methods shall be used to support tests or in lieu of tests whenever practical to satisfy verification requirements.

- o The verification program will confirm that hazards identified by FMEA or other analysis have been eliminated by design or reduced to an acceptable level using safety devices, warnings or special procedures.

- o The planning of the verification program shall provide for flexibility to accommodate changes necessitated by verification results, program redirection, or as a result of continuous evaluation/monitoring of the cost/value effectiveness of verification activities.

- o After each flight, minimum testing will be performed consistent with determining that refurbishment, repairs, and reconfiguration were correct and that the system is ready for reflight. In general, testing for the next flight may be limited to that

required to validate refurbishment, repairs, and configuration changes made after the previous flights.

- The policy regarding test documentation requirements at various management levels shall be flexible with the objective of minimizing the variety, quantity and formality of documentation required.

5.1.7.3 Verification Approach

The verification approach presented herein is a baseline approach identified during the course of the study. It is based on the protoflight hardware build and test concept used successfully by GSFC, is compatible with STS philosophy and follows the guidelines and criteria stated previously. In essence the approach does not deviate substantially from past approaches used for spacecraft type programs. As discussed previously, some reduction in scope may be possible for later flights based on experience and data gained from early missions.

Since verification by test is by far the costliest method, the following discussion will generally be limited to test activities. It is assumed that analysis and other assessment methods are used in parallel to support test activities or are used independently to verify other appropriate characteristics.

Figure 5.1.7-1 is the verification program general flow. It shows the end-to-end sequence of AMPS instrument, Labcraft and complete payload verification. The flow has two major parts, namely; instrument and Labcraft design and development and complete payload integration and checkout. The two are joined through a milestone designating flight certified status of all equipment entering the integration cycle. Before discussing the individual elements, the following observation can be made regarding the allocation of verification costs and potential areas of cost reduction.

The instrument and Labcraft design and development phase is more demanding on program resources than is the integration phase. It is also more flexible since the instrument development is done on an individual basis allowing the use of custom made development programs for each instrument consistent with its role during the mission. Therefore, the opportunities for potential program economies rest with individual instrument project management during its design and development. The integration and checkout phase will be much more rigidly constrained and offers fewer opportunities to affect program economies.

The discussion of instrument and Labcraft design and development is subdivided into two parts; (a) component verification and (b) individual system certification.

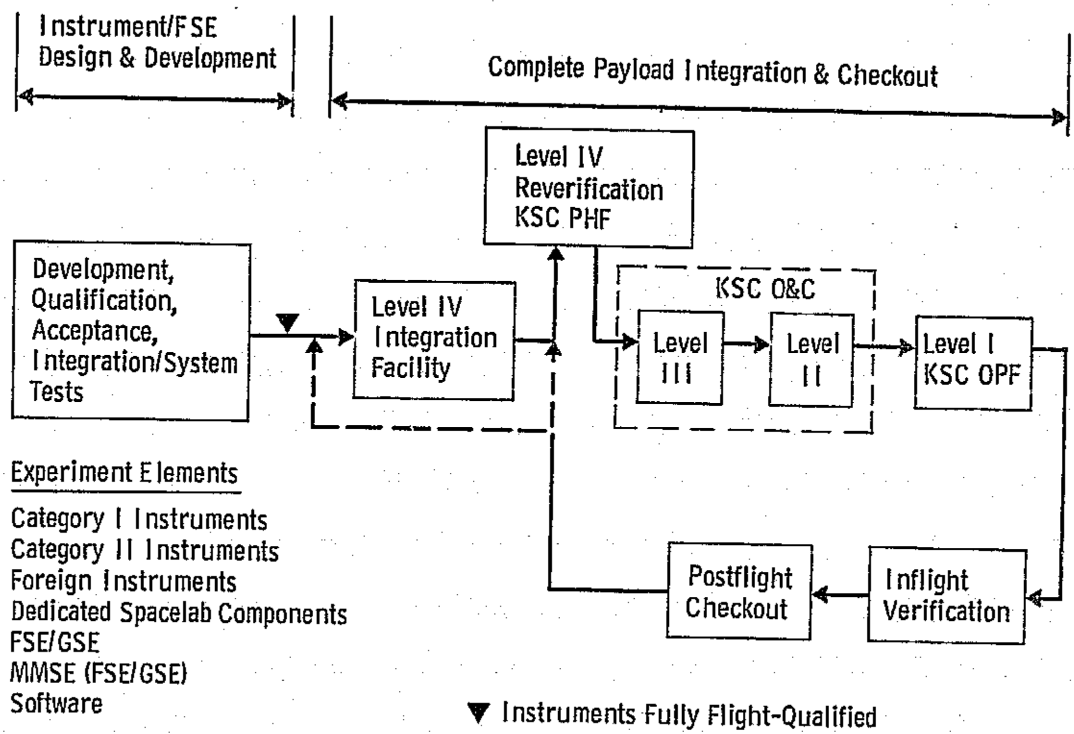


Figure 5.1.7-1 Verification General Flow

Component Verification - The verification flow during the instrument and Labcraft component design and development is shown in Figure 5.1.7-2. It is generalized to accommodate the verification requirements of components which are diverse in nature and development status. The term "component" as used herein encompasses items better known as "black boxes" and also subsystems. The flow is the same for instrument and Labcraft type equipment, therefore, unless distinction is made, the following discussion pertains to both types.

The center of the figure in heavy outline emphasizes the protoflight build and test concept. The thrust of the concept is to build, test, refurbish and fly the same article. The components, therefore, will undergo a series of classical qualification tests to ensure reasonable success during system level testing and successful flight(s). The design of such a qualification test program must balance many factors to achieve satisfactory level of confidence, yet not to overtest the articles. As indicated in the figure, project management must weigh the overall test exposure and such factors as cost, design features and associated history, mission objectives, operational mode and environments. Modifications after test failures and refurbishment after test completion, if necessary, will be part of the plans.

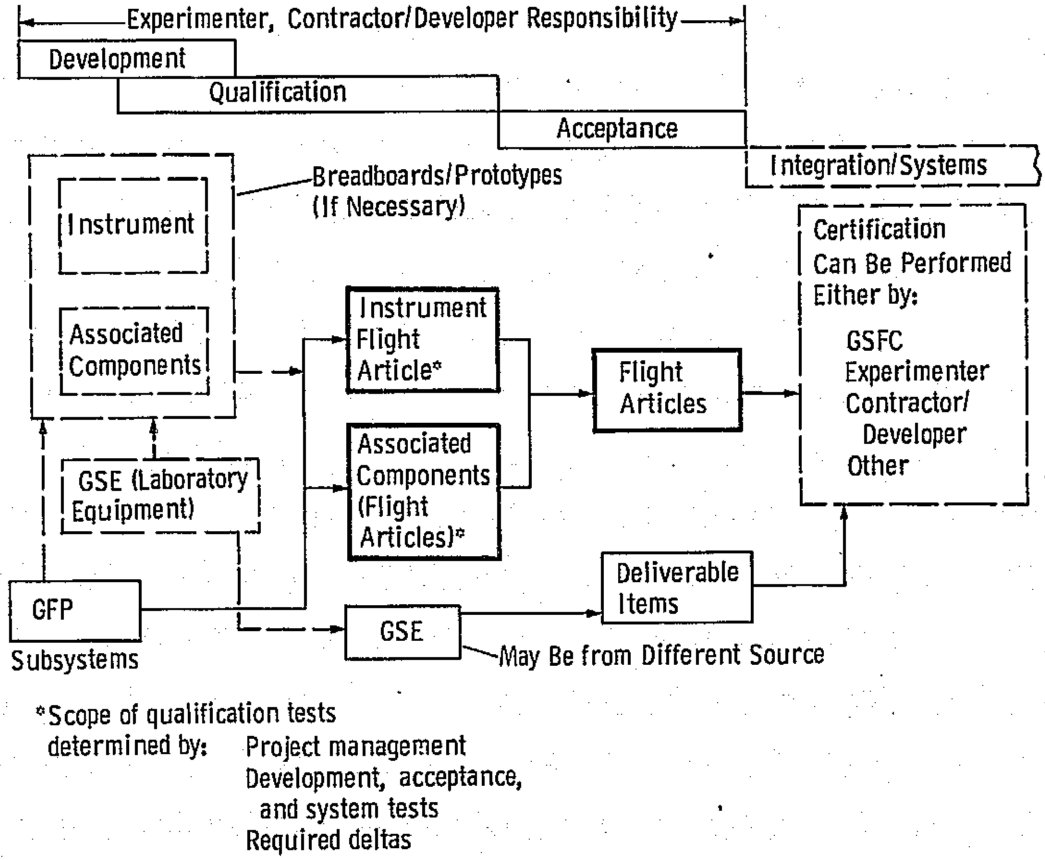


Figure 5.1.7-2 Component Verification

Figure 5.1.7-2 also shows a longer development path for items requiring additional development testing prior to protoflight article build. These tests will use breadboard/brassboard/prototype articles in a laboratory environment. Test configurations will include off-the-shelf standardized hardware as well as laboratory type support equipment. Successful tests during this phase will allow a decrease in qualification testing. It is expected that most Labcraft equipment will go through the prototype stage of development and testing. This equipment will typically be built in excess of one unit and their use by instruments require valid performance and reliability baselines.

Acceptance tests at component levels will be used either for quality/workmanship screening, establishing of functional baselines before qualification tests, after refurbishment and prior to integration in a higher level assembly. The exact use of acceptance tests for any one item allows much latitude in selection of applicability and use of environments. Following component level tests the components will be integrated in their respective higher level assemblies for system level tests and certification.

Individual System Certification - The instrument level test flow is shown in Figure 5.1.7-3. Typically this is a higher level of assembly which includes the components discussed in the preceding paragraph, standardized hardware, support equipment and software. The assemblies will represent a functional instrument entity. This assembly and test phase for various instruments may take place in several locations depending on programmatic considerations and existing capabilities. The objective of this activity is to integrate the instrument functional elements and to subject the flight system to a series of environmental and special tests. These tests and other previous verification activities designated as requirements for certification will complete the certification cycle.

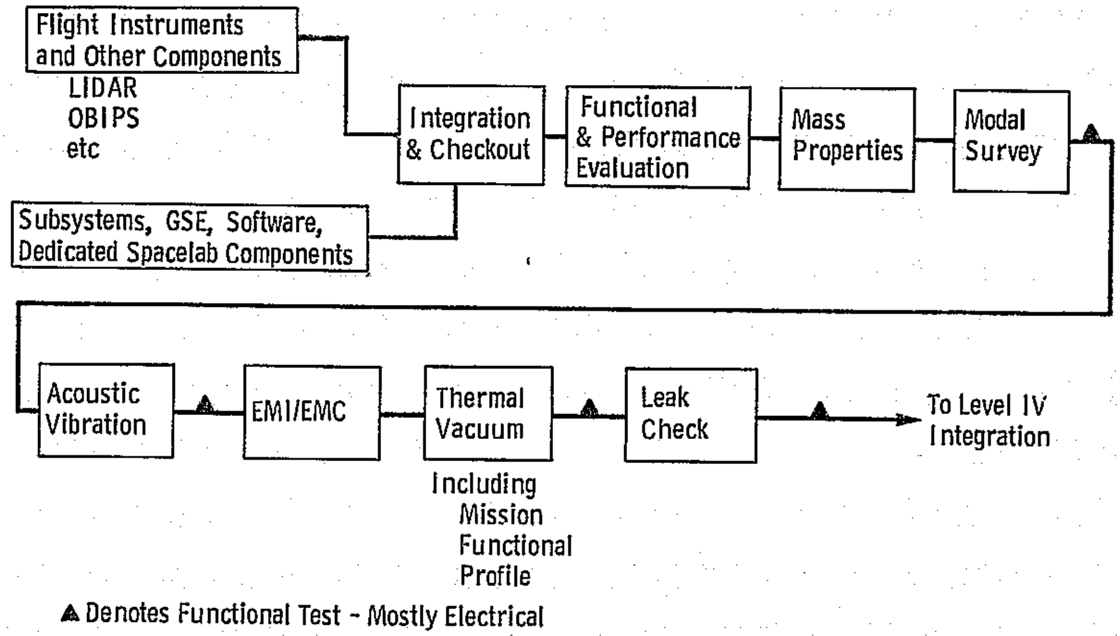


Figure 5.1.7-3 Individual System Certification

Figure 5.1.7-3 shows a typical series of tests which may not be required for all instruments. Here again, the project management must choose the applicable tests in light of similar factors as those for component qualification tests. Additions or modifications may be necessary for some instruments, i.e., added magnetics evaluation or thermal test in lieu of thermal vacuum. The test phase will start with integration and functional checkout followed by functional and performance evaluation. The latter will include system parametrics as well as evaluation of system sensitivities. The results will serve as a functional baseline for determination of effects from subsequent environmental exposures. After final test the instrument will be subjected to a thorough functional test in preparation for shipment to the integration site.

5.1.7.4 Complete Payload Integration and Checkout

Following the instrument system level tests the AMPS payload elements will begin the complete payload integration cycle. It will take place in several levels progressing from instrument, Labcraft and pallet integration (Level IV) to Spacelab/AMPS payload and Orbiter (Level I). The successive levels emphasize the integration and checkout of new interfaces associated with the new level of integration. It should be noted that verification objectives decrease with each successive level of integration and the activities become more operations oriented. Refer to the verification general flow (Figure 5.1.7-1) for integration sequence and relationships.

Level IV Integration - The objective of Level IV integration is the integration and checkout of individual instruments, pallets, racks, GSE, Labcraft, simulations and the complete AMPS payload (soft-mated). It is to be performed at the prime contractor's site. Since it is the first and lowest level of integration it will be more detailed and extensive in scope. Consequently, from a verification point-of-view it will satisfy many requirements. Figure 5.1.7-4 is a functional flow of Level IV integration for the first AMPS payload. It shows a gradual buildup at the individual instrument level leading to complete payload configuration integration and checkout. Besides functional verification it will also include first time evaluation of EMI/EMC at the complete payload level. As indicated in the figure, the flow will be significantly reduced for follow-on flights with the elimination of pallet level tests (acoustic vibration and modal survey). It is considered necessary to perform these tests on the first set of pallets to acquire data for confirmation of analysis and modeling results. The tests will be performed with a single pallet at a time in a facility other than the clean room used for integration and checkout. To accomplish this, the pallets will be demated after complete payload tests and returned to the same configuration and functional status afterwards. Next, the pallets will be demated, prepared and shipped to KSC.

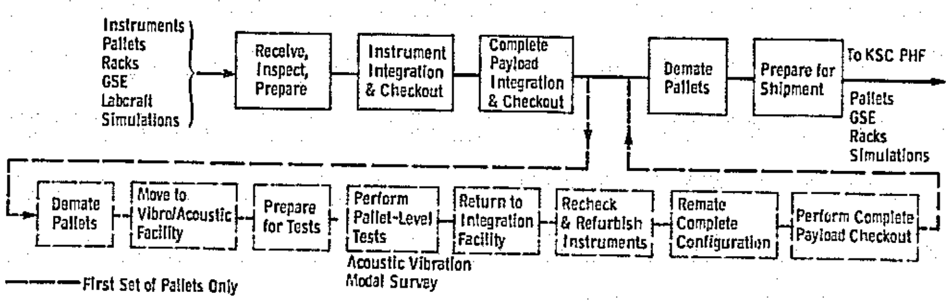


Figure 5.1.7-4 Level IV Integration Flow

The assumption regarding the pallet level tests is that the entire payload complement, i.e., instruments and Labcraft are to be flown for the first time. If, however, a significant number of Labcraft or instruments will have been flown previously, the necessity for all or part of pallet level tests should be reconsidered. Another factor to enter this decision will be the availability of applicable data from previous Spacelab flights, i.e., orbital flight tests.

Figure 5.1.7-5 is included to show the functional configuration of Level IV integration. It shows the instruments on a pallet interfacing with the data bus and their own unique GSE. This dual interface is desirable for gradual integration, troubleshooting, and evaluation of the science data interface not accessible through the data bus. The computational equipment in combination with peripherals will perform the functions and simulations of the Spacelab and Orbiter systems not part of this configuration. Software used by this equipment will be, as far as possible, Spacelab and instrument flight software modified for ground use. Simulations will be substituted for missing functions and interfaces. After the completion of Level IV integration at the prime contractor's site the pallets will be demated and transported to KSC Payload Handling Facility (off-site).

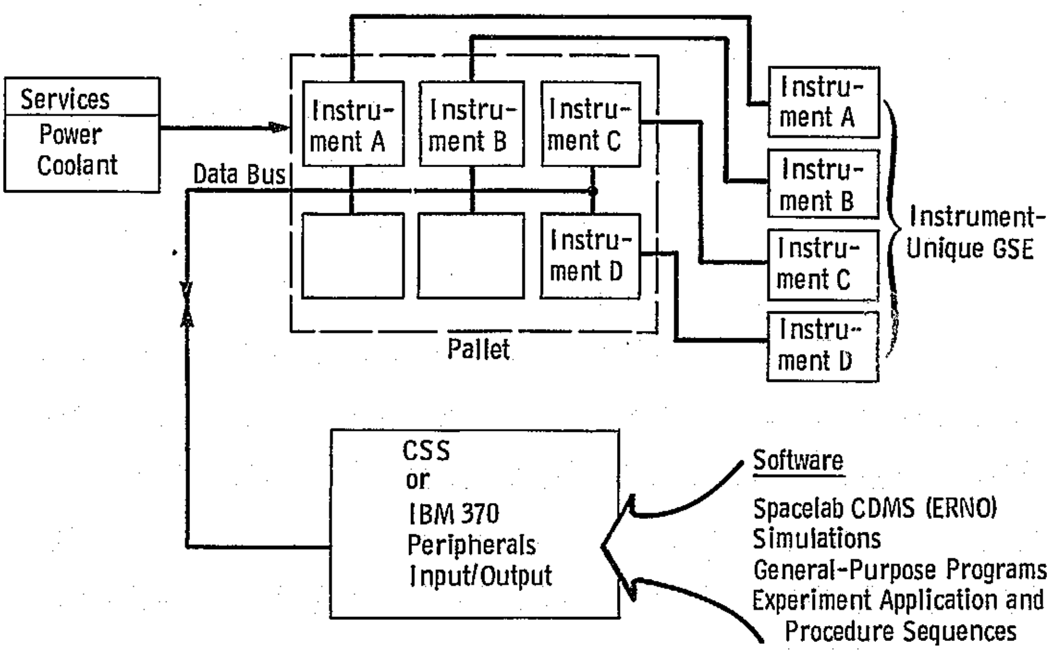


Figure 5.1.7-5 Level IV Integration Configuration

Level IV Reverification - The objective of reverification of the Level IV configuration at KSC is to reconfirm the functional status of the payload which might have been altered due to elapsed time and effects of transportation. It is also likely that some changes may be

necessary prior to committment for further integration. To achieve this the pallets will be remated (soft) and, using the same support equipment configuration shown in Figure 5.1.7-5, brought up to final complete payload functional status which existed at Level IV integration.

It should be noted that this reverification activity will be the last phase under payload development center control. Therefore, it is the final opportunity to perform certain types of final checkout which may be time consuming or may require special conditions or equipment. From the PHF the payload pallets will be transported for Level III/II integration and checkout in the Operations and Checkout Facility.

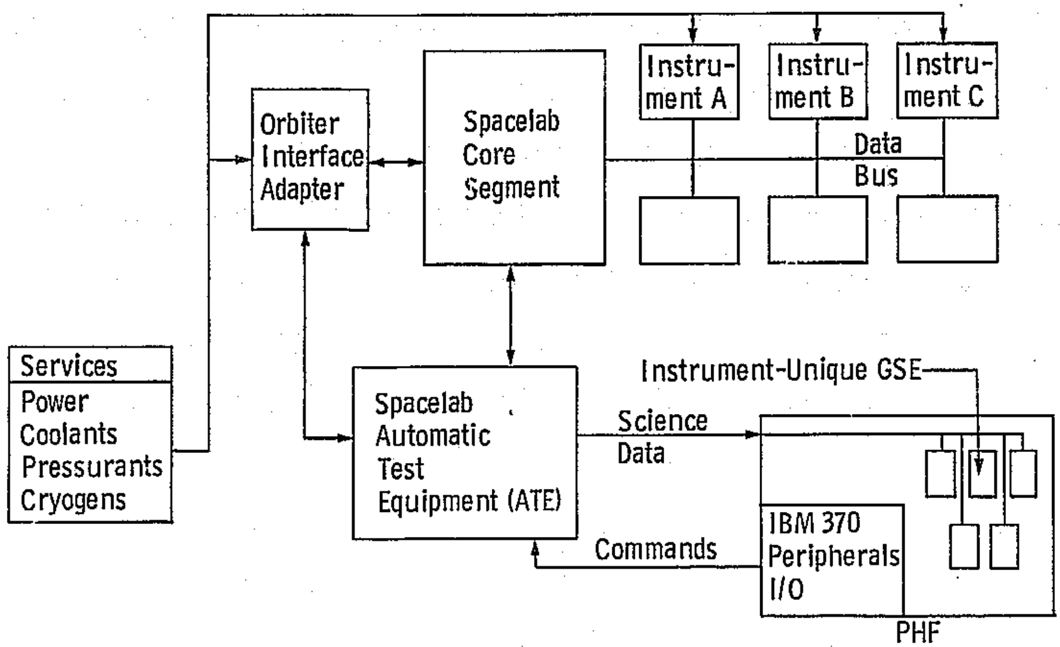
Level III/II Integration - The objective of Level III/II integration is the integration and checkout of the pallet train, racks, Spacelab and complete Spacelab/AMPS configuration. The Spacelab and payload will be assembled in the integration and checkout stand and mated to support equipment. Figure 5.1.7-6 shows the functional configuration of the airborne and ground equipment. New in this configuration as compared to Level IV integration is the actual Spacelab Core Segment with its Automatic Test Equipment (ATE) and Orbiter Interface Adapter (OIA). Also shown is a tie-in of previous configuration support equipment, located at the PHF, with payload via the ATE. The instrument unique GSE located at the PHF can only receive science data demultiplexed by the ATE while other support equipment can command the instruments.

This activity will see the hard mate of pallets, installation of racks and step-by-step integration and checkout of Spacelab with the payload. The features and functions to be verified include physical accommodations, utility services, command and data mangement and software. After the confirmation of overall functional compatibility several system level tests will be performed. These are: mission simulation, EMI/EMC, determination of science data rate capabilities and data interface with MCC and POCC. Weight and c.g. determinations will be made as part of handling of the Spacelab and payload assembly for transferring to the OPF which is the location for Level I integration.

Level I Integration - The objective of Level I integration is the integration and checkout of the remaining new interfaces. They are:

- o Spacelab to Orbiter physical interfaces (fit, clearances) and functional interfaces (power, coolant, command and data, caution and warning and Launch Processing System).
- o Tunnel installation involving fit and leak tests.
- o Payload Specialist Panel installation involving physical fit and functional tests.

Following the verification of individual interfaces, integrated system checkout of Orbiter/Spacelab interfaces will be performed as part of the Orbiter Integrated Test. During this test AMPS participation will be in a passive support role of providing required functions and responses. AMPS will play a similar role during rollout, final checkout at the pad and launch.



Software

Spacelab Subsystem } Including AMPS Requirements
Spacelab CDMS }
Orbiter Interface Simulation
Instrument Applications and Procedure Sequences (Ground Version)

Figure 5.1.7-6 Level III/II Integration Configuration

5.1.8 Environments

The environments used to perform preliminary design and definition of the AMPS Flight 1 payload will be discussed under the following four topic headings:

- o STS-natural and induced;
- o Electromagnetic Interference/Compatibility;
- o Spacecraft Charging;
- o Contamination.

Shuttle Transportation System Natural and Induced Environments - The natural and induced environments utilized for the AMPS preliminary design and definition of the pallet and Spacelab pressurized module mounted elements were those defined in:

- o Space Shuttle Systems Payload Accommodations, JSC 07700, Vol XIV, Rev D, Change 16; and

- o Spacelab Payload Accommodations Handbook, ESTEC, SLP/2104, May 1976.

Electromagnetic Interference/Compatibility - Figure 5.1.8-1 shows a functional schematic of the total Orbiter/payload electromagnetic environment analysis task that is required for Labcraft payloads. To date only portions of the "static magnetic" and "dynamic electric" fields have been evaluated. Although the analytical capability and skilled personnel exist to complete the analysis there is a major deficiency in the required input data. As a minimum the parameters listed in Table 5.1.8-1 are required to do an adequate EMC analysis for AMPS.

The only data now available is some preliminary Orbiter data. To complete the task our recommendations are:

(1) EMI measurements should be made at appropriate stages during the ground based systems test sequences on Orbiters 101 and 102. The objective of these measurements is to determine the Orbiter EMI characteristics such that their contribution to the overall environment of the payload can be determined. The measurements should address the following:

Electric Field Strength Measurements

- o Identify and characterize discrete sources--type, power, etc.

- o Establish rate of decrease with distance

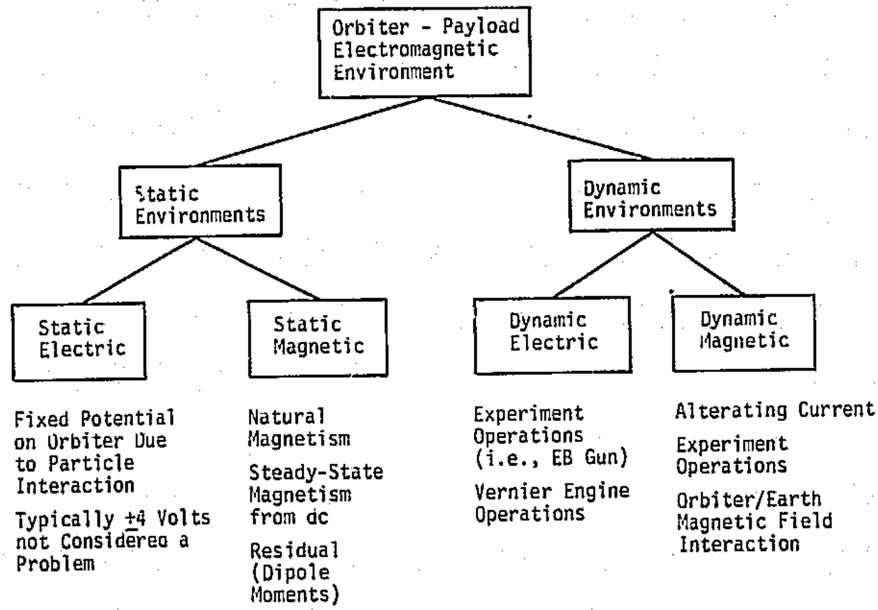


Figure 5.1.8-1 Electromagnetic Environments

Table 5.1.8-1 Required Parameters

Objectives Experiment Measurements Accuracy Instrument Capability Range Sensitivity Bandwidth Signal to Noise	Instrument Threshold Power Spectral Density Broadband Narrowband Orbiter/Spacelab Dipole Moments Emitters Timelines Typical EM Emissions Etc.

- Use instrumentation that will provide narrowband measurements
  - 50 to 300 Hz bandwidth
  - Low noise figure amplifiers
  - Slow scan rate
- Relate measurements to frequency analysis for data to characterize discrete source characteristics
- Measure EMI at manipulator arm max locations
- Change frequency limits to 30 Hz to 3 GHz for narrow band and 14 KHz to 1 GHz for broadband

Magnetic Field Strength

- Measure AC magnetic field--30 Hz to 25 MHz as with electric field
- Separate remnant and stray field measurements
  - Characterize remnant field of vehicle with power off, but after system test
  - Characterize magnetic dipole strength of major Orbiter subsystems
  - Model vehicle by using these characterized subsystem characteristics
  - Determine stray sources by probing with magnetometer and gradiometer during system test
  - Characterize stray sources

(2) Determine Spacelab contributions by performing similar ground measurements during systems test

(3) Determine instrument/systems/experiment characteristics such as:

Instrument Susceptibility

- Broad bandwidth
- Discrete frequencies
- Power system noise
- Conducted noise

Instrument Measurements

- o Frequency range of measurement
- o Anticipated signal strength
- o Required signal to noise ratios
- o Accuracy requirements on geophysical field measurements

Instrument Emanations

- o Instrument compatibilities
- o Instrument/system compatibilities
- o Operational schedule constraints

(4) Develop an universal EMI/EMC computer model which predicts the integrated EMI environment of any given Labcraft payload based on the discrete emitting sources identified in 1, 2 and 3 above.

(5) Conduct on-orbit measurements to validate model, and to determine distance fall-off characteristics in a plasma environment.

A survey was made of existing instrument designs appropriate for making on-orbit measurements of the EMI field (item 5 above) in the vicinity of the Orbiter payload bay. The best candidate instrument identified is the combination of two receivers developed for the Mariner Jupiter Saturn mission. One of these instruments in the Planetary Radio Astronomy Receiver (PRAR) (P.I., Dr. James Warwick, University of Colorado), and the other is the Plasma Wave Subsystem (PWS) (P.I., Dr. Don Gurnett, University of Iowa). The PRAR covers the frequency range from 40 mHz down to 1.2 kHz with a dynamic range of 140 dB. The PWS extends the coverage down to 10 Hz, and overlaps the PRAR up to 56.2 kHz. These instruments have been designed to operate in close proximity and to share a common antenna forming an efficient and compact integrated instrument. The fact that they are designed for the space environment makes it a simple matter to adapt them to an RMS deployment mode to carry out Orbiter EMI mapping.

Spacecraft Charging - The spacecraft charging analysis indicated that a high current ($\sim$1 amp) accelerator could not operate for any significant ($\sim$1 sec) length of time, in orbit because the free electron density at 210 kilometers cannot provide adequate compensating return currents to the Orbiter. Further complicating the problem of neutralization is the construction of the Orbiter which is covered with extremely low-conductivity ceramic tiles required for thermal

considerations. As an isolated body the total capacitance of the Orbiter is equivalent to 10^{-9} farads. Therefore, small net charge imbalances result in high Shuttle voltages with respect to the ambient plasma.

Due to the analytical difficulty of this problem the approach recommended to answer both the charging and safety questions is:

(1) <u>For Early Flights</u>

- Use moderate current ($\sim$1 amp) electron beam source
- Operate the beam at increasing current levels
- Include instrumentation to monitor:

  Ambient electron density
  Spacecraft voltage relative to plasma
  Charge density collected on sample dielectric coverings

(2) <u>For Later Flights</u>

Develop high-current electron and ion beam instruments as a single instrument;

Operate both sources simultaneously to neutralize net charge emission.

<u>Contamination</u> - This subject is discussed in detail in Section 5.1.9.

5.1.9 Contamination

The AMPS contamination analysis investigated the contamination characteristics of the Orbiter, Spacelab and AMPS equipment and established preliminary ground and mission design and operational requirements that would assure proper payload operation and the return of usable scientific data from space. The analysis was performed in four steps as listed below:

- Identification of contamination sources and their generic effects;
- Identification of contamination sensitive instruments and equipment;
- Performance of a detailed contamination effects analysis; and
- From the analysis, provide recommendations, solutions and conclusions.

Contamination Sources and Effects - Contamination of sensitive systems and instruments occur during manufacturing, validation, launch-to-orbit insertion, on-orbit, re-entry and landing, and post-flight operations. Of these six activities, the on-orbit activity is recognized as the most significant and by far the most difficult to control. The study concentrated on identifying the sources and effects during that period of activity.

Listed in Figure 5.1.9-1 is a summary of the identified major contamination sources for the Shuttle Orbiter and the Spacelab carrier as well as their generic effects on sensitive instruments. Figure 5.1.9-2 shows the Shuttle Orbiter/Spacelab configuration and the contamination source locations. In addition to the carrier contamination sources shown, certain AMPS type instruments and equipment also tend to increase contamination. These were also reviewed and identified. Those instruments along with the severity of the problem is shown in Figure 5.1.9-3.

Contamination Sensitive Instruments - Instruments most susceptible to contamination are those having cryogenically cooled optics, requiring long exposures, making measurements predominantly in the direction of the velocity vector and those instruments making measurements in the X-ray or ultraviolet (UV) portion of the spectrum. Figure 5.1.9-4 shows the results of our analysis for the AMPS Flight No. 1 instrument complement.

Sources

Evaporator
- Water Vapor and Ice Particles

Thruster Exhausts (VCS, RCS, OMS, SRB, and Separation Rockets)
- Various Gases, Particles

Outgassing (Nonmetallic Materials)
- Various Light and Heavy (Polymer) Molecules

Leakage (Cabin Atmosphere)
- N, O, Biological Byproducts, Food and Hygiene Products.

Particle Release
- Ground Deposits, Thrusters, Evaporator, Thermal Expansion and Contraction, High-Energy Radiation, Micrometeoroids

Return Flux (Interaction with Ambient and Self)

Effects

Deposition on Critical Surfaces
- Absorption, Scatter, and Reradiation of Radiant Energy
- Obscuration and Diffusion of Target Images
- Discoloration

Induced Atmosphere
- Absorption, Scatter, and Reradiation of Radiant Energy
- Reduced Image Contrast from Bright Background
- Alteration of Ambient Atmosphere
- False Stars, Streaks, IR Energies

Figure 5.1.9-1 Summary of Contamination Sources and Effects

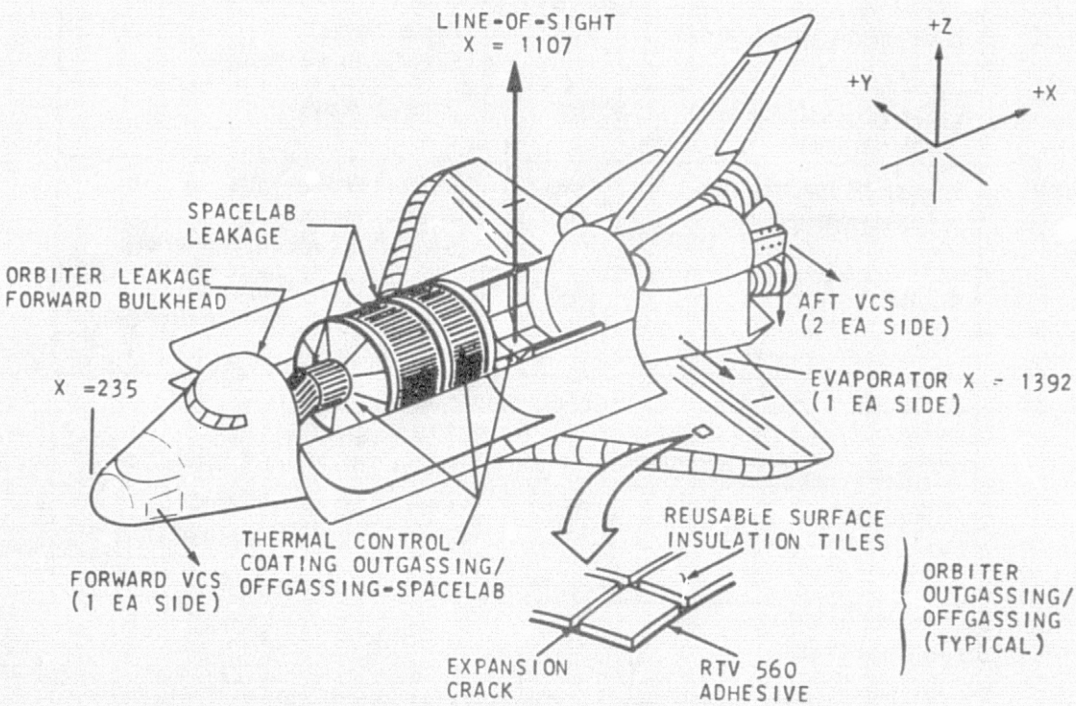


Figure 5.1.9-2 Shuttle Orbiter/Spacelab Configuration and Contaminant Source Locations

Instrument	Threat and Process
Maneuverable Subsatellite	Minor from Propulsion Gas Emission
Electron Accelerator	Minor to Moderate from Shuttle Charging
Ion Accelerator	Minor to Moderate from Shuttle Charging and Ion Species Emission
Plasma Accelerator	Minor to Moderate from Species Emission
Meteor Gun	Minor from Propulsion Gas Emission
Relativistic Electron Accelerator	Minor from Simultaneous Plasma Emission
LLLTV-SS	Minor from Propulsion Gas Emission
Cryo Cooling Systems Venting	Minor Contribution to Mass Column Density

Note: When these instruments are operating the instruments listed on Figure 5.1.9-4 should not be operated.

Figure 5.1.9-3 AMPS Instruments Which Tend To Increase Contamination

Flight	Instrument	Susceptibility	Basic Reasons
No. 1	1. Cryogenic Limb Scanner	High	Cooled Optics Long Daytime Exposure
	2. Far IR Spectrometer/ Interferometer	High	Same as Above
	3. Near IR Spectrometer/ Interferometer	Low	Medium Wavelengths Warmer Optics Short Exposure (1/2 Nighttime).
	4. LIDAR	Low	Medium Wavelength Medium Exposure (Nighttime Only) Baffled Optics
	5. OBIPS	Low	Medium Wavelength Short Exposure
	6. Solar Flux Calibration Package	High*	Very Short Wavelengths Absolute Values Required
		Low**	No Contaminant Access

*Telescope External to Slit
**Telescope Internal to Slit

Figure 5.1.9-4 AMPS Instruments Contamination Susceptibilities

5.1.9.1 Contamination Analysis

Figure 5.1.9-5 presents the overall functional flow of the contamination analysis process and Figures 5.1.9-6, 5.1.9-7, 5.1.9-8, and 5.1.9-9 indicate some of the typical "Contamination Transport Characteristic" results that are used in our analysis. These results were calculated using our "Contamination Computer Model" and employing the types of input data as depicted within the dotted line of Figure 5.1.9-5. The closed form analytical model approach used for this study was shown on Skylab to be an effective tool in contamination evaluation and assessment. An analysis of this nature allows the basic parameters to be identified, geometric considerations to be established and formulates in a systematic perspective the trends that would evolve from variation of the important physical parameters which affect the contaminant susceptibility of sensitive experiments to be flown on AMPS.

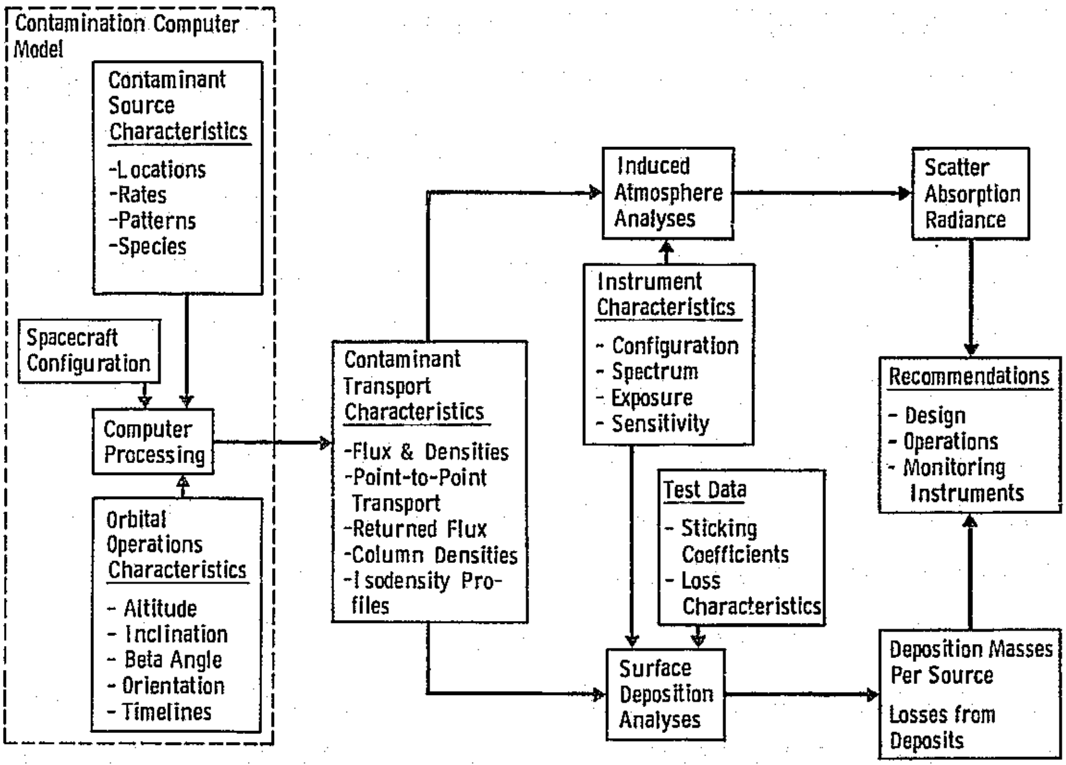


Figure 5.1.9-5 Contamination Analysis Processes

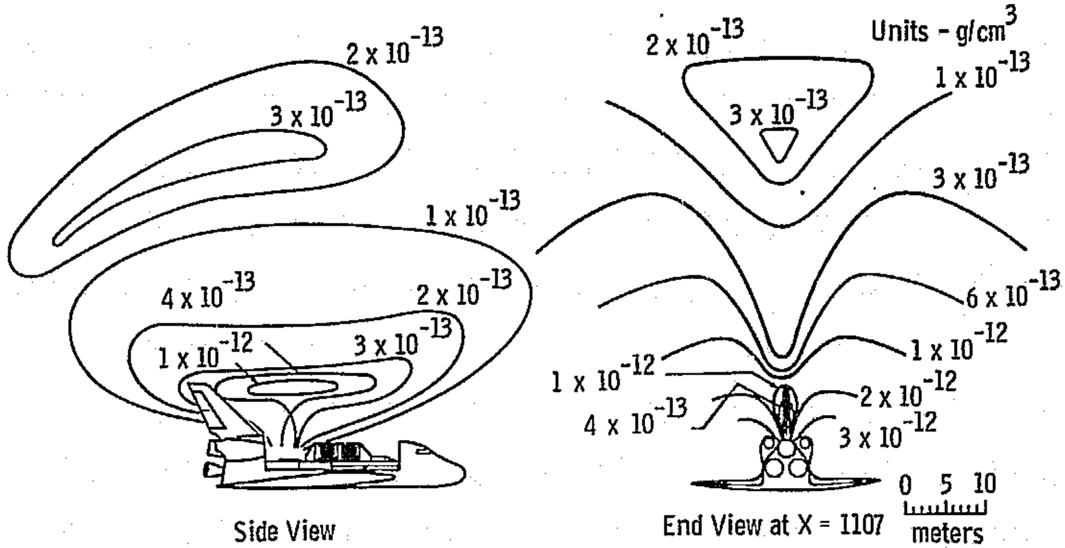


Figure 5.1.9-6 Supplemental Flash Evaporator Isodensity Contours

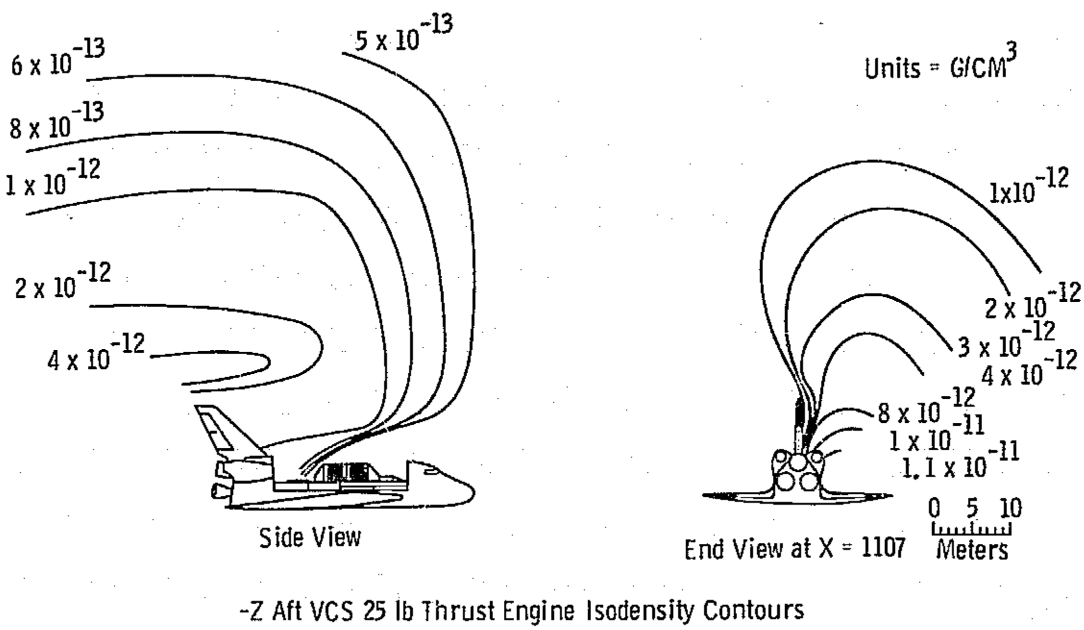


Figure 5.1.9-7 -Z Aft VCS 25 Lb Thrust Engine Isodensity Contours

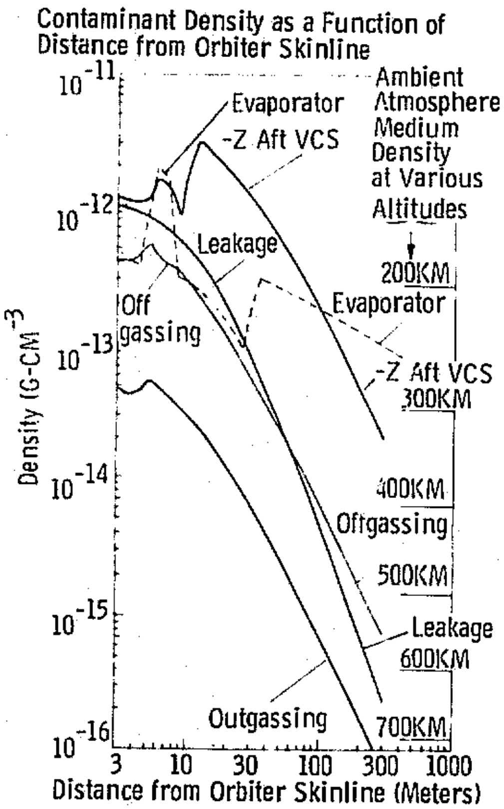


Figure 5.1.9-8 Shuttle/Spacelab Contaminant Density Characteristics

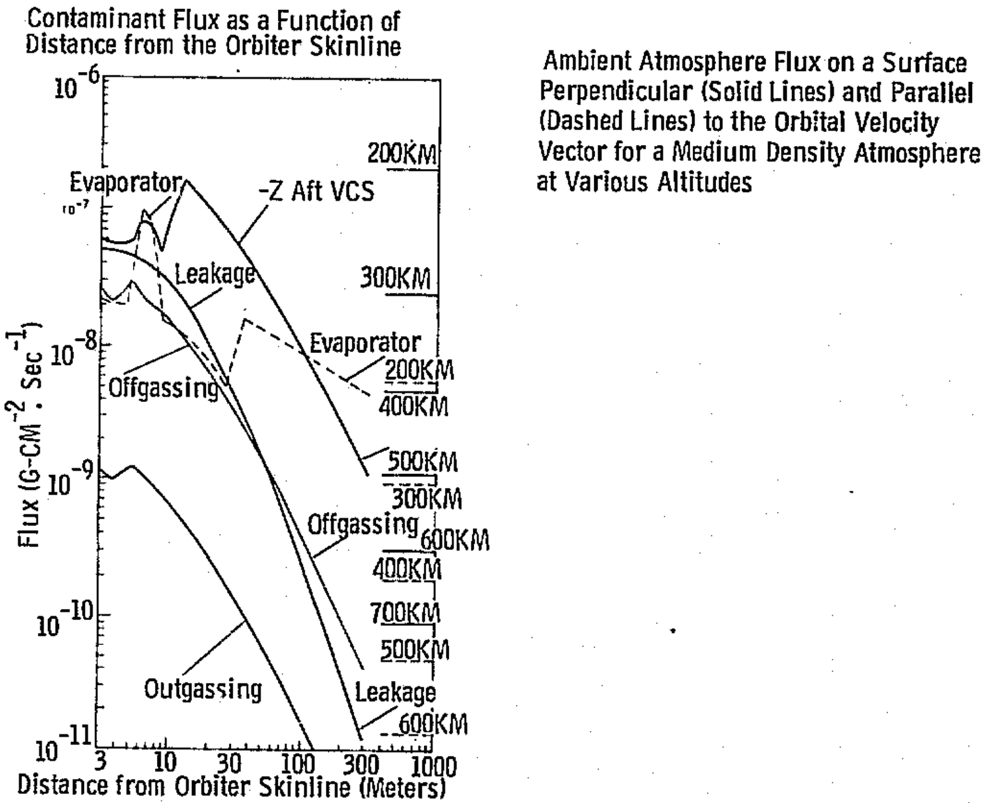
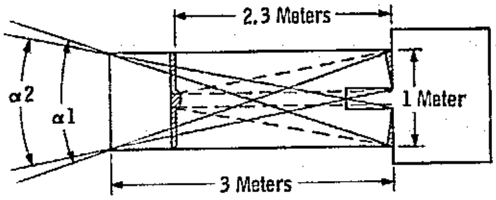


Figure 5.1.9-9 Shuttle/Spacelab Contaminant Flux Characteristics

As mentioned, a review was conducted of all the AMPS flight instrumentation. Those instruments which were designed particularly susceptible to contamination were analyzed in detail against the contaminant induced environment as developed through our computer analysis. Results of several of the instrument analyses are shown in Figures 5.1.9-10, 5.1.9-11 and 5.1.9-12 and have been summarized in Figure 5.1.9-4. Also shown on the following figures are the assumptions and estimated operation cycle used in the analysis for each instrument, the calculated losses and the recommendations to minimize the contamination effect on the instrument.



Estimated Losses	Percentage, End of Mission
-20°C Optics (λ = 0.5μ)	
Velocity Vector	65
All Directions	22
0°C Optics (λ = 0.5μ)	
Velocity Vector	2
All Directions	0.7
0°C Optics (λ = 0.35μ)	
Velocity Vector	8
All Directions	3
+20°C Optics (λ = 0.5μ)	
Velocity Vector	1
All Directions	0
+20°C Optics (λ = 0.35μ)	
Velocity Vector	4
All Directions	1.4

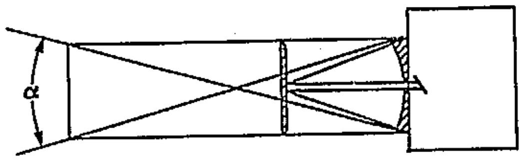
Assumptions and Operations

1. Configuration as Shown
2. Wavelength = Visible Spectrum
3. Cover Used When Not Operating
4. Optics Temperature Variable
5. Total Exposure Time = 50 Hr (Nighttime)
6. Instrument Pointed so that Only Contaminant Return Flux Enters Baffle.
7. Orbit Altitude = 210 km
8. Present Orbiter and Spacelab Contamination Characteristics

Recommendations

1. Operate at Higher Altitude Where Return Flux is Much Less.
2. Extend Light Baffle.
3. Heat External Optics.
4. Delay Operations Until Spacecraft Has Cooled (~5 min)

Figure 5.1.9-10 LIDAR--Contamination Loss



Estimated Losses	Percentage, End of Mission	
	Day	Day/Night
Light Baffle Walls Cooled		
Velocity Vector	46	25
All Directions	17	9
Light Baffle Walls Uncooled		
Velocity Vector	95	78
All Directions	55	33

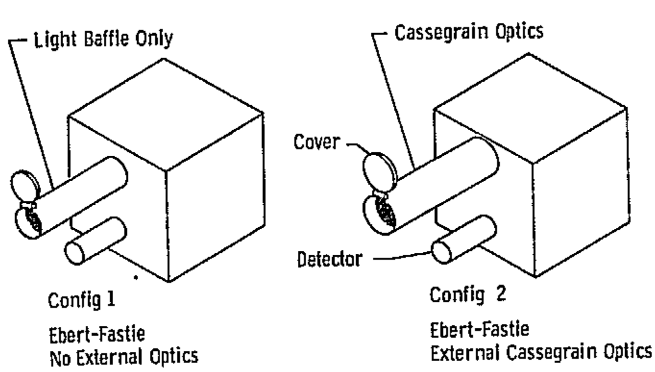
Assumptions and Operations

1. Aperture = 30 cm
2. Light Baffle Length = 120 cm
3. Wavelength = 5-50 Microns
4. Cover Used When Not Operating
5. Optics Temperature = 20°K
6. Total Exposure Time = 86 hr (Day/Night)
7. Instrument is Pointed so that Only Contaminant Return Flux Enters Baffle
8. Orbit Altitude = 210 km
9. Present Orbiter and Spacelab Contamination Characteristics

Recommendations

1. Operate at Higher Altitudes
2. Extend Light Baffle
3. Do Not Operate Evaporator
4. Point Instrument Away from Velocity Vector.
5. Expose Optics Only During Early Portion of Orbit Daytime (e.g., First 20 Minutes).

Figure 5.1.9-11 Far IR Interferometer/Spectrometer--Contamination Loss



Losses	Percentage, End of Mission	
Config 1	Essentially Zero	
Config 2	At λ = 1100 Å	32
	At λ = 3500 Å	2.2

Assumptions and Operations

1. Configurations as Shown
2. Acceptance Angle - 0.19 Steradians
3. Exposed Toward Earth Limb 50 Hours/Mission, Orbit Nighttime
4. No RCS, VCS, or Evaporator Operation
5. External Optics Near 0° C
6. Instrument Pointed All Directions But Such That Only Return Flux Can Enter
7. Orbit Altitude = 210 km
8. Present Orbiter and Spacelab Contamination Characteristics

Recommendations

1. Operate at Higher Altitude.
2. Heat External Optics.
3. Extend Light Baffle
4. Delay Operations Until Spacecraft Has Cooled
5. Use No External Optics

Figure 5.1.9-12 UV Spectrometer/Photometer

5.1.9.2 Conclusions and Recommendations

Many of the AMPS instruments are sensitive to contamination and will require special attention both during design and operation. Particularly those instruments making measurements in the very short wave length region and those having very cold optics. It may also be necessary to inhibit some of the Orbiter and Spacelab venting and operations of certain systems prior to and during the operation of sensitive instruments to prevent excessive degradation to the data.

The recommendations from the study are listed below:

- Prepare a detailed AMPS contamination control plan as shown in Figure 5.1.9-13;
- Perform detailed contamination analysis covering both ground and mission for the payload (systems and instruments);
- Implement system and instrument design and operational requirements and constraints early to reduce program impacts and cost; and
- Provide real time on-orbit contamination measuring instruments as shown in Figure 5.1.9-14.

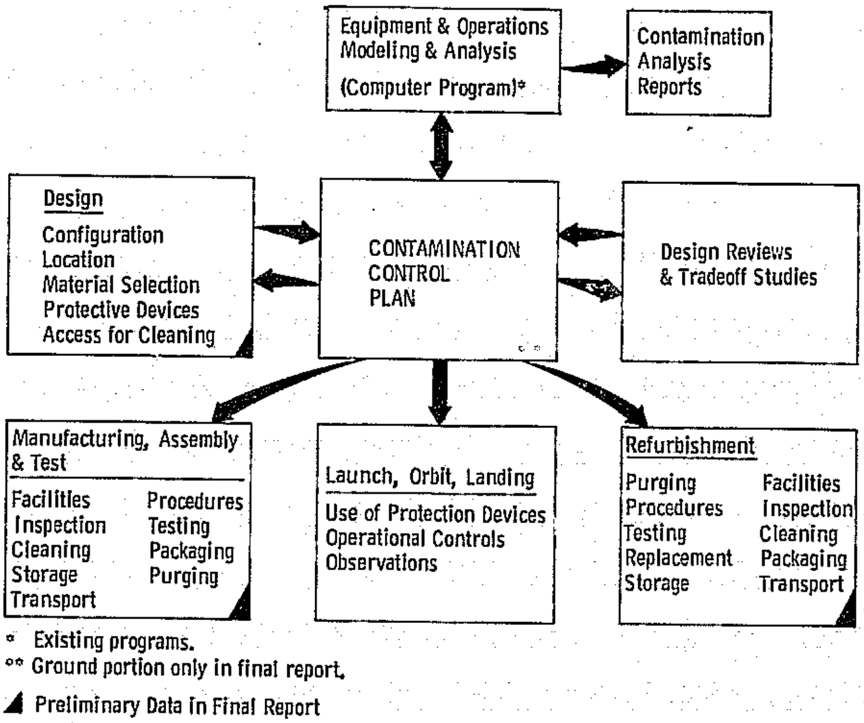
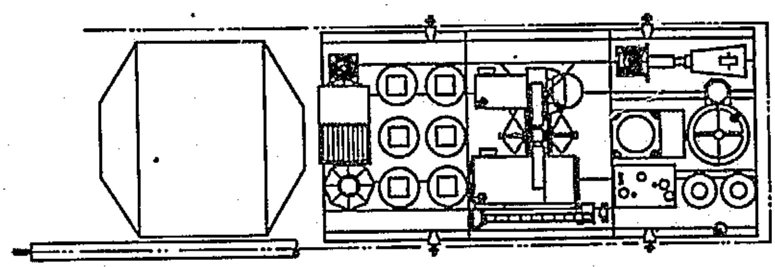


Figure 5.1.9-13 Contamination Control Plan Development



Recommended

TQCM

Mounted on Telescope of Extremely Susceptible Instruments
Same Acceptance Angle
Near Same Temperature, Where Possible
At Least 10Å Deposit Sensitivity
Real Time Readout

Photoelectric Photometer

Mounted on Pointing Platform
Protected, and Used Briefly
Real Time Readout

Calibration Sources

On Susceptible Instruments
Protected, and Used Briefly
Real Time Readout
Use Complete Optical Train

Justification

Provides Operational Control.
Provides Specific Data Correlation
Provides Basis for Select Instrumentation for Future Missions Where IECM is not Carried
Essential on Extended Missions

● = Calibration Source
▭ = TQCM
⇎ = Photoelectric Photometer

Figure 5.1.9-14 Recommended Contamination Monitoring Instruments

5.2 Structures and Mechanisms Subsystems

5.2.1 Design Integration Trade Offs

This section covers the approach and rationale utilized for indivdual instrument location within the AMPS pallet train. Major location drivers such as weight, size, center-of-gravity (CG), field of view (FOV), coalignment requirements, and operational interfaces are addressed.

5.2.1.1 Requirements/Constraints

There are specific AMPS requirements and constraints that must be considered when laying out a Spacelab pallet payload. These constraints result from Orbiter and Spacelab design configurations and include; volume, weight, CG and hardpoint locations. Volume constraints include the overall Orbiter payload bay envelope (15 ft. dia. by 60 ft) and the volume limitations incurred by using the pallet. The payload launch capability of 65,000 pounds (29,484 kg) has never been a constraint for the AMPS type payloads in that the abort return capability of 32,000 pounds (14,515 kg) must be satisfied. The Spacelab three pallet train load capability of 11,023 pounds (5,000 kg) has been a constraint in that the AMPS flight one launch weight is 12,130 pounds (5,502 kg). Alternate pallet configurations such as a separated pallet train (one pallet + two pallets) do provide an increased load capability. A single pallet can accommodate 6,874 pounds (3,118 kg) and a two pallet combination limit is 11,023 pounds (5,000 kg). As in all small module plus three pallet payloads, the longitudinal center of gravity location becomes the most important constraint and the most difficult to meet. This is apparent by noting that the forward located module and tunnel along with their outfittings make up 33% of the AMPS payload weight. In addition, the forward CG limit is 11.5 inches (2.86 metres) behind the module centerline for a 32,000 pound (14,515 kg) payload. Y and Z axes CG limits have not been a serious design problem. A majority of the AMPS instruments are in the weight range that requires attachment to pallet hardpoints. Experience with the AMPS payloads has shown the location and quantity of hardpoints cause payload layout difficulties. Because of this, instruments must share pallet hardpoints and this leads to interface control and integration problems. As a design groundrule, no instrument mounting across a pallet splice was permitted. This self-imposed constraint was to prevent payload integration problems and to eliminate racking loads between pallets.

5.2.1.2 Discussion

The initial task in developing the flight arrangement was to determine the design drivers and the order of instrument placement. For Flight 1 the significant instruments (because of weight) are: the Electron Accelerator (I-9) including Pulse Power Supply, the LIDAR (I-1), and the Gas Release (I-21). Other significant instruments because of pointing requirements are: OBIPS (II-3), Cryo Cooled Limb Scanner (II-7), Near IR Spectrometer (II-9), and the Cryo IR Spectrometer (II-10). (Section 5.10 defines the physical characteristics of the instruments.)

All have pointing requirements that necessitate gimballed pointing platforms. A study of pointing system options was pursued to define the best system(s) independent of pallet location.

Three pointing systems considered during this trade were the Miniaturized Pointing Mount (MPM), Instrument Pointing System (IPS) and the Small Instrument Pointing System (SIPS). No new designs were considered in keeping with the goal of maximum usage of multi-mission support equipment. A comparison of the four instruments along with the three pointing platforms is summarized in Figure 5.2.1-1. The goals of this study were to achieve an integrated pointing system that met pointing requirements and achieved the following: minimize volume occupied, minimum modifications, and minimize new Labcraft.

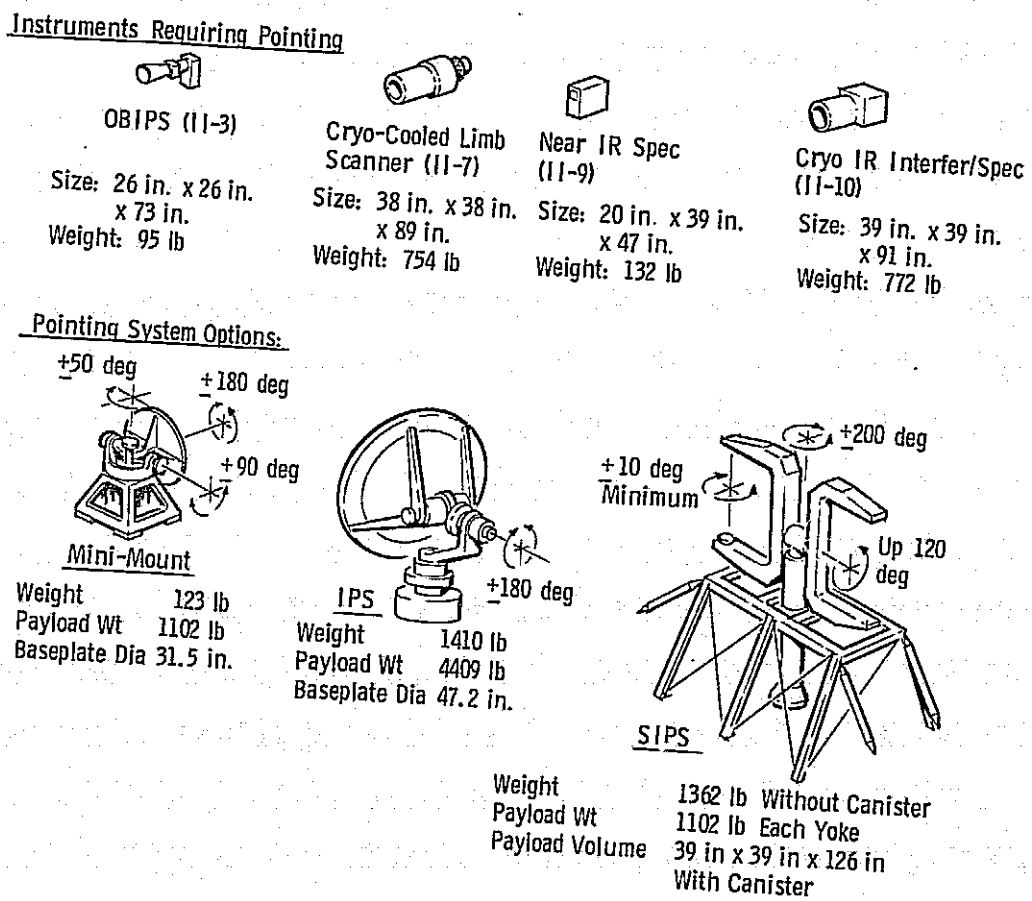


Figure 5.2.1-1 Comparison of Instruments and Pointing Systems

In addition to pointing capability, over-the-sill viewing (limb scanning) and coalignment of the cryo instruments (II-7 and II-10) are required. The Near IR Spectrometer requires environmental conditioning to maintain proper operating temperatures. A self-contained canister on the pointing platform was assumed similar to the multi-mission support equipment (MMSE) environmental canister or the SIPS heat pipe canister. Table 5.2.1-1 lists the basic trade study rationale. Advantages and disadvantages of the various platforms as far as meeting the above requirements are presented along with the selected system.

Table 5.2.1-1 Pointing Platform Trades/Rationale

REQUIREMENT: COALIGNMENT BETWEEN II-7 & II-10 (MOUNT TOGETHER)

Options	Advantages	Disadvantages
MPM	o Minimum weight & vol.	o Platform capability 1102 lb vs. 2205 lb combined weight (1) o Coalign fixture required capable of adjustment - costly and complex (2) o Launch/landing lock strut system required to instrument CG (new Labcraft) (3) o Additional mounting structure required for over sill FOV
IPS	o Has good payload weight to platform weight ratio	o Same as (1) above o Same as (2) above o Same as (3) above
SIPS	o On orbit alignment possible using sensors between instruments o Launch/landing locks included as part of platform o Normal extension allows for sill FOV o Instruments size compatible with yoke.	o Takes up most of pallet o Requires new interface hardware between yoke and instruments

Table 5.2.1-1 (Continued)

REQUIREMENTS: II-9 REQUIRES ENVIRONMENTAL CONTROL

Options	Advantages	Disadvantages
IPS		o New canister design or mod SIPS o New design launch/landing lock strut system required o Additional mounting structure for over sill FOV
SIPS	o Normal extension allows over sill FOV o Launch/landing lock system exists o Canister design exists	o Takes up most of pallet (even if combine II-3 with II-9 too much space lost) o Too much capability for weight of II-9 (1321 lb vs 2204 lb permissible)
$\boxed{\text{MPM}}$	o MMSE canister design exists o Minimum weight & volume o Launch/landing lock strut & mechanism design existing	o Requires additional mounting structure for over sill FOV

Table 5.2.1-1 (Concluded)

REQUIREMENT: II-3 REQUIRES POINTING PLATFORM (REMAINING INSTRUMENT

Options	Advantages	Disadvantages
IPS		o Too much capability for size of instrument o Requires additional structure for over sill FOV
SIPS	o Normal extension allows over sill FOV	o Too much capability for size of instrument. o Takes up most of pallet
MPM	o Minimum weight and volume o Best payload to platform weight ratio	o Requires additional structure for over sill FOV

As shown in the table, SIPS was selected for supporting the cryo instruments. The primary reasons for SIPS selection were the basic dimensional compatibility of the instruments with the SIPS yoke and the independent operation of the yokes, which allows on-orbit alignment by sensors. A single MPM was eliminated due to inadequate load capability. Two MPM's would solve the weight problem and allow coalignment through the use of sensors between the instruments. However, the added complexity of new launch/landing lock structure and additional structure for over-the-sill viewing negates this approach. The MPM system was chosen for mounting of the Near IR Spectrometer because of the existence of a canister design and the minimum space used. IPS would require a new or modified canister plus new Labcraft, and it would not be an effective use of the SIPS capability. The selection of the MPM for OBIPS was based on minimum pallet space usage. Because of the low instrument weight, the additional design and build task to allow over-the-sill FOV would be minimal.

The next step in the payload layout task was the actual positioning of instruments within the pallet train. The aft pallet is critical as far as achieving the longitudinal CG and instruments were located there first. Figure 5.2.1-2 presents a weight comparison of the major Flight 1 instruments, Labcraft and deployed modules. The electron accelerator (with pulse power supply) and LIDAR were selected for the aft pallet because

they share the pulse power supply (operational interface) and their combined weight is high. The 1323 pound (600 Kg) pulse power supply was located on the floor of the pallet and the accelerator and LIDAR were mounted on top of the power supply to minimize cable runs. The accelerator was located forward on the pallet to prevent beam interference with the aft bulkhead and because the LIDAR receiver is heavier. The two LIDAR transmitters require bore-sighting with the receiver, and were located on the side of the aft pallet on a single platform to facilitate alignment.

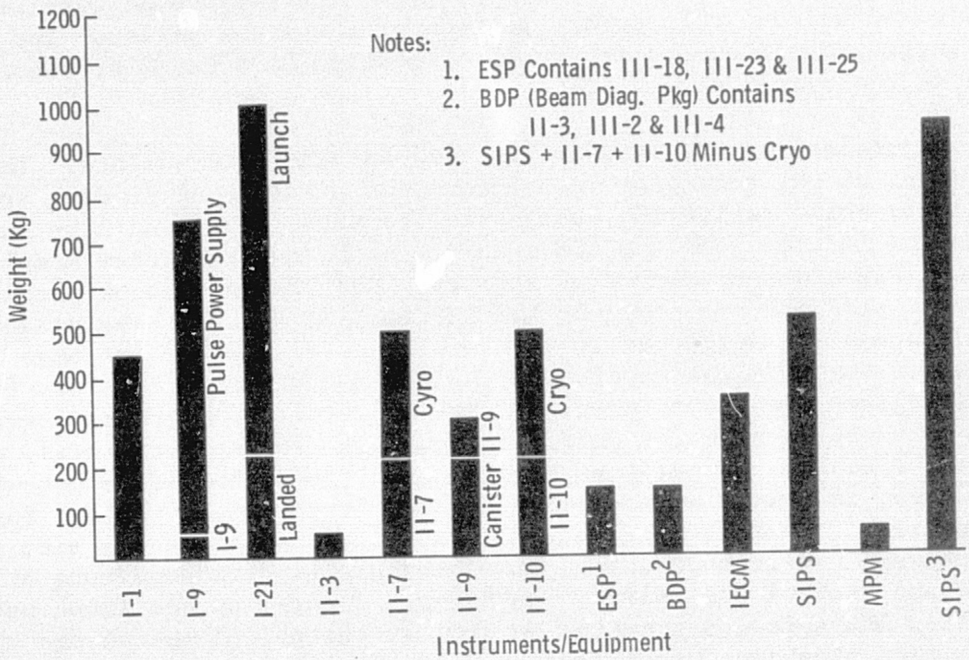


Figure 5.2.1-2 Instrument Weights for Flight One

From Figure 5.2.1-2 it can be seen that SIPS with the two cryo cooled instruments was the next priority item to be positioned. In keeping with locating the heavy items aft, the 3366 pound (1527 Kg) SIPS was located on the center pallet. The SIPS support structure interfaces with 10 pallet hardpoints at the middle of the pallet. This leaves 14 hardpoints available, 7 in a line on each of the front and back edges. This arrangement effectively eliminated the center pallet for consideration as a location for any of the other large or heavy instruments without sharing the SIPS hardpoints. A check of the CG shift due to emergency jettison of SIPS and its instruments was made after the total payload configuration was completed. The analysis revealed a forward CG shift of 19.0 inches (.48 metre) for the abort/return case. This left a CG margin of approximately 27.5 inches (.7 metres).

Planned ejection of the gas release modules results in a reduced return weight of 2425 pounds (1100 kg) launch and 502 pounds (228 kg) landed weights including Labcraft). The aft end of the forward pallet was selected for the gas release modules. A platform type support structure with truss members to the pallet hardpoints was used to mount the six gas release modules. The near IR spectrometer in the environmental canister (on the MPM) has a stowed length and width of 90 inches (2.28 metres) and 40 inches (1.01 metres) respectively. A requirement to coordinate viewing with the cryo-cooled instruments meant that Y axis viewing was needed. This dictated a transverse pallet mounting position because of the MPM gimbal range ($\pm 90^{\circ}$ and $\pm 50^{\circ}$). A review of the available space on the pallets at this time eliminated all but the forward portion of the front pallet. The MPM was supported off the pallet in this location by a truss structure to achieve the over-the-sill FOV.

Major instruments were now located on all three pallets. The placement of the remaining instruments was a matter of finding space available while satisfying design requirements. The remaining instruments and modules were the ESP module, the beam diagnostics module, OBIPS, IECM and the solar flux monitor. Requirements to be met included a position reachable by the RMS for ESP and the beam diagnostics package and OBIPS had to look over the sill for gas release viewing. The IECM was located on the aft pallet because it was the heaviest item and fit an opening forward of the LIDAR transmitters. This left the +Y side for OBIPS which was mounted near the sill. Since the beam diagnostics package is over 116 inches (2.95 metres) in length, it had to be mounted along the length of a pallet. Space was available near the sill of the center pallet on the RMS side and the package was located there using separate support structure at the front and back of the pallet. The ESP was located on the front pallet to allow RMS reach. A position on side of the pallet was chosen where deployment clearance was adequate.

5.2.2 Mechanisms

The purpose of this study is to document the rationale used to determine the design approach for the AMPS mechanisms. Past experience has shown the mechanisms task to be a major design and test effort. Because new mechanism designs involve complex interactions between material selection, tolerances, load paths, lubricants and basic design concepts; an AMPS guideline was established to use tried and proven systems whenever possible. A cursory review of the Flight 1 instruments was performed to determine the obvious mechanism requirements. From this analysis, identified mechanisms included; ejection device for the gas release modules (I-21), separation devices for the ESP and the gas release modules, and latch/unlatch devices for the deployed modules (ESP and beam diagnostics package). Further design and payload layout effort identified additional needs for launch/landing locks, spin systems, deployment devices, and emergency jettison systems.

5.2.2.1 Requirements

Table 5.2.2-1 is a listing of the AMPS Flight 1 mechanisms, and also provides functional description, general design requirements, alternate design concepts, and identifies the instrument or module the mechanism is used with. Based on experience, the list of design alternates was abbreviated to the most feasible approaches.

Table 5.2.2-1 AMPS Mechanisms - Function, Requirements and Design Alternates

MECHANISM	FUNCTION	REQUIREMENTS	DESIGN ALTERNATES	REQUIRED ON INSTRUMENT	REMARKS
Capture & Release Device	Provide multiple latch & unlatch operations	Withstand launch/landing loads. Accommodate misalignment provide connector mate/demate.	Hook & latch*	ESP Beam Diagnostics Package	*Use existing design
Ejection System	Impart a velocity & direction to an ejected item	ΔV within tolerance Minimize tipoff Reliability Contamination considerations	Springs Thrusters Gas pressure	I-21 ESP	
Launch/Landing Locks	Provide structural support for item operated in a different position than stowed position	Withstand launch & landing loads. Accommodate misalignment	Hook & latch	II-3* II-9* III-2	*Locks on pointing platform. III-2 sensor requires launch/landing restraint
Separation Devices	Release an item for ejection or deployment	Single release Reliability Minimize shock loadings	Separation nuts Band clamp Ball locks Linear shaped charge	I-21 ESP	
Spin System	Impart rotational velocity to item/instrument	Rotational velocity within tolerance Reliability Spin about principle axis	Gear motor Thruster Gas release	ESP	
Emergency Jettison	Separate an item that has malfunctioned in a deployed position	Single Release Reliability Safety	Separation nuts Ball lock Linear shaped charge Band clamp	I-1 II-3 II-7 II-9 II-10	I-1 covers, II-3 & II-9 MPM, II-7 & II-10 exists as part of SIPS
Deployment Device	Extend/Retract probes, antennaes and sensors	Reliability	Telescoping booms Hinged booms Storable tublar extendable members (STEM)	III-2 III-22	

5.2.2.2 Discussion/Selection

The selection process used for each class of mechanisms involves a description of the alternate concepts, identification of design requirements and key functional requirements, and evaluation of advantages and disadvantages for each alternative. Key functional requirements used as rating factors included; reliability, safety, performance and comlexity.

The capture/release device and the launch/landing locks are discussed together as their requirements and operations are similar. Another pair of mechanisms that have common design requirements and are combined for discussion purposes are the separation and emergency jettison devices. Here the requirements and functions are identical, but separation is planned while jettison is needed only under malfunction circumstances.

Capture/Release and Launch Locks - This mechanism has the potential for use on a number of Shuttle payloads. Many planned experiments or instruments are deployed by the RMS during operation and restowed for return. Other instruments require an operating position other than the stowed position. These instruments all require a mechanism capable of providing launch support, multiple release and attach operations, and alignment guides for capture operations. In addition, the capture/release device must have provisions for mate and demate connectors.

A literature search of mechanisms capable of performing these tasks was initiated rather than looking at new design concepts. MMC has just finished a study task for MSFC which evaluated mechanisms for the "Integrated Orbiter Servicing Study" (MCR-75-310, Contract NAS8-30820). In this study effort, 12 interface mechanisms were compared and rated. From these results, two designs were completed utilizing the best features of the studied mechanisms, and engineering models of these designs were fabricated for proof of concept. One of these designs was intended for a bottom interface while the other was for side mounting. The bottom mount design satisfied all of the AMPS needs. The basic design concept is the use of a motor driven gear system to rotate a hook linkage which engages rollers in receptacle pins. Two pins are provided along with outrigger pads for reacting side loads. Figure 5.2.2-1 depicts the configuration. The only identified problem for AMPS usage is the relatively small misalignment capability. A modification would be required to allow capture under conditions involving greater offsets.

AMPS Flight 1 requires launch lock devices on the two MPM platforms to isolate the gimbals from launch and landing loads. For this usage, design requirements similar to those on the capture/release devices apply with the following difference. The amount of misalignment to be accommodated is reduced because of the programmed stowage sequence of the MPM. The other launch lock requirement is to support launch loads on the vector magnetometer sensor. Here, sensor deployment away from the magnetometer electronics package is needed for proper operation. For the launch/landing lock devices, a similar design to the capture/release device would be used with modifications to produce a more compact design.

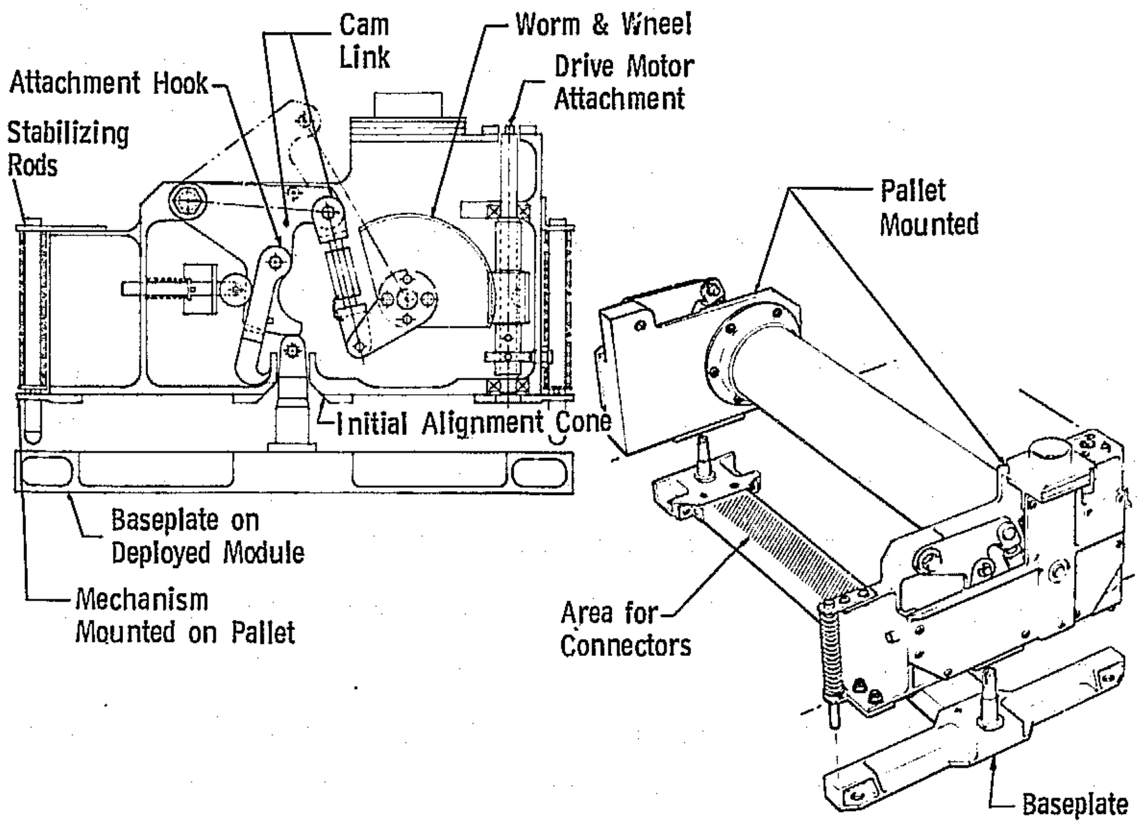


Figure 5.2.2-1 AMPS Mechanisms - Capture/Release Device

<u>Ejection System</u> - An ejection system provides a specified delta velocity between an object separated from the Orbiter and the Orbiter. The specific needs for AMPS Flight 1 involve the gas release modules and the ESP module. These two packages present an interesting design case in that the ejected weights are similar but the ΔV requirements are .35 metres/second for the ESP and 5.0 metres/second for the gas release modules.

Design alternates investigated were a spring system, thrusters and a gas pressure system. The spring system makes use of multiple compression springs to force bodies apart. The thruster system utilizes solid propellant rockets to provide the separation forces. The gas pressure scheme uses gas released from a storage vessel to drive a piston attached to the ejected body. A listing of the three approaches along with advantages and disadvantages of each is provided in Table 5.2.2-2. Some of the data used for ratings was taken from "Flight Separation Mechanisms" (NASA SP-8056). The spring concept receives the highest rating

for low momentum systems. Analysis shows a required average spring force of 26 pounds (114 newtons) and 7760 pounds (34,520 newtons) for the ESP and gas release modules respectively (using a 2.5 inch stroke). It can be seen that there is a potential ground safety problem when the ejection forces exceed the weight of the ejected portion. A spring system was chosen for the ESP module and eliminated from consideration for the gas release module. The use of thrusters was not considered for the gas release module because of the contamination potential. A thruster system would be feasible for the ESP as the ejection takes place from the end of the RMS, however the spring system is less complex. The gas pressure scheme was chosen for the gas release module for two reasons. First the alternate design compliments the spring design by exposing the design and manufacturing problems that become evident in the preliminary design effort. Also, the gas system requires less average force because a longer stroke is possible.

Table 5.2.2-2 Ejection Systems - Design Alternates

RATING FACTORS	SPRINGS	THRUSTERS	GAS PISTON
Simplicity	Good	Good	Poor
Reliability	Good	Good	Good
Contamination	Good	Poor	Good
ΔV prediction	Fair	Good	Good
Weight	Good for small ΔV & weight Poor for high weight & ΔV	Good	Poor for low weight & ΔV Good for high weight & ΔV
Safety	Good	Fair	Good
Remarks	Match springs to minimize tipoff & achieve more accurate delta V	Require alignment for proper eject direction	Require alignment & matching of piston force-time history

<u>Separation and Emergency Jettison Devices</u> - Use of separation devices on AMPS include the release of the gas release modules and the ESP module at the time of ejection. Emergency jettison is required on contamination covers that violate the payload clearance envelope when deployed. Jettison is also required on the pointing platforms to provide release if a malfunction occurs in a deployed position. This discussion covers both separation and jettison devices as a single topic using the term "separation" for both systems.

Separation devices have been in use since the earliest space flights and much documented experience exists on these devices. As in other areas, no attempt was made to develop new concepts, but rather to use proven flight qualified devices. The choice of available designs includes separation nuts, band clamps, ball locks, cable cutters and linear shaped charges. Separation nuts are an explosive device where a charge separates the nut from a bolt and a piston impact ejects the bolt. These devices can be non-fragmenting and non-contaminating by the use of a sealed mechanism. Band clamps consist of a V-band that fits over mated flanges on two bodies and is held in place by band tension. Point release devices are used in several places on the band to provide redundant release. Ball locks are ball headed bolts held in place by a lock sleeve which is released by a gas or solenoid operated piston. A linear shaped charge is an explosive encased in a metal sheath which when detonated cuts through structure. Table 5.2.2-3 lists the design alternates along with a simplified rating of the functional characteristics required for adequate operation. Separation nuts have the highest rating in the table and were chosen for use on AMPS ejected instruments. The emergency jettison designs were not fully developed, but separation nuts seem to be the logical choice again. The use of band clamps or ball locks as alternate schemes require more investigation on actual jettison designs.

Table 5.2.2-3 Separation and Jettison Devices - Design Alternates

RATING FACTORS	SEPARATION NUTS	BAND CLAMPS	BALL LOCKS	LINEAR SHAPED CHANGE
Load Capability	Good	Good	Good	Good
Simplicity	Good	Good	Poor	Good
Safety	Good	Fair	Good	Fair
Contamination	Good	Good	Good	Fair
Shock loading	Good	Fair	Good	Poor
Reliability	Good	Good	Good	Good
Remarks	High reliability when used with redundant initiators	Must restrain bands, possible field joint, joint preload hard to predict	Requires close tolerances, requires solenoid/thruster to activate	High shock loading, can produce severe contamination

Spin System - A low speed (4 RPM) spin system is required to provide spin stabilization for the ejected ESP module. The ESP also requires rotation while deployed on the RMS. Here rotation at the same speed is used to provide proper orientation of the peripheral mounted sensors. Use of the ESP module several times each mission requires a spin system with a repeat capability. Reliability, spin accuracy, and elimination of wobble are other requirements the spin system must satisfy.

Design alternates evaluated were solid propellant thrusters, gear motor drive, and gas pressure. The thruster concept uses tangential firing, solid propellant, rocket motors to provide the spin up torque. In the gear motor scheme, an electric motor drives a gear train that is attached to the rotating portion. The gas pressure concept is similar to the thruster concept except that cold gas is released through a nozzle from a pressure vessel. All these approaches use a dry lubricated, bearing-mounted, spin table with a lock arrangement to prevent spin until on-orbit activation. The gear motor approach is the only one that is normally reusable. By sequence firing of groups of thrusters or by providing an adequate storage reservoir, the other concepts could provide respin capability. These modifications add complexity and weight and reduce reliability. Advantages of the selected gear motor system include rotational accuracy and reliability. Disadvantages include high weight and complexity.

Deployment Device - These devices are used to extend and retract antennas, probes, and sensors. AMPS Flight 1 instruments requiring deployment devices are; vector magnetometer sensor, langmuir probe, and the dipole antenna on the ESP module. Deployment distances of 8 inches (.2 metre) to 40 inches (1.0 metre) are required for these instruments. General requirements the deployment devices must satisfy include simplicity, reliability, low weight, and, for some applications, high stiffness. Reliability is important when used on opposing antennas on a spinning ejected module such as the ESP. Stiffness is a requirement for the vector magnetometer sensor deployment device because of the weight of the sensor and the position stability requirements.

Alternate designs considered were telescoping tubes, hinged tubes, and storable tubular extendible members (STEM). In the telescoping tube concept, sliding concentric tubes are extended to provide a rigid boom. The hinged beam design uses linked tubular members that are folded for storage. Storable tubular extendible members consist of a tape or element stored on a drum. The tape assumes a tubular shape when extended. Table 5.2.2-4 lists the advantages and disadvantages of the design alternates. The selection of the STEM design is based on using the qualified existing design even though it normally has a greater deployment length than is needed. The ability of a STEM or bi-STEM to provide the proper stiffness and stability for the magnetometer sensor has not been fully investigated. For this application, some additional study is needed to determine whether an alternate design such as the telescoping tubes would be a better choice.

Table 5.2.2-4 Deployment Devices - Design Alternates

CONCEPT	ADVANTAGES	DISADVANTAGES
Telescoping Tubes	o Simple o Reliable o Stiff	o Space requirements - not compact. o Stability/stiffness is dependent on tolerances. o Cold welding, galling could be problem. o New design
Winged Tubes	o Simple o Reliable	o Drive & locking becomes more complex when require retraction. o New design
Storable tubular extendible member	o Space qualified o Reliable	o More capability than need o Lower stiffness

5.2.3 Dynamics and Vibroacoustics

The broad spectrum of experiments and payloads planned for Space Shuttle, coupled with the differences between the Shuttle and current expendable launch vehicles, presents new problems and concepts in structural dynamics. Since the maximum steady-state acceleration of the Shuttle is only approximately 3 g's (limit), the low frequency dynamic loads imposed by transient and quasi-sinusoidal events such as liftoff, gusts, thrust termination, POGO, and landing may become increasingly important in the design of primary structure. In addition, continually increasing requirements for pointing accuracy and alignment for experiments dictate the need for associated improvements in analytical methodology and design techniques.

The current predictions for acoustic levels in the payload bay are higher than those of current launch vehicles. Unless reduced, this environment could have significant impact on the cost of AMPS/Labcraft type payloads both in the design and the test phases.

There have been a number of recent studies and model programs (references 1, 2 and 3) which address these problems with the ultimate goal of reducing shuttle program costs. Some of the results and concepts developed under these studies are included in the recommended approach to the structural dynamics program for AMPS, described in the following paragraphs.

Vehicle Dynamics -The basic analytical tool used for solving various dynamic problems either in the time or the frequency domain, is FORMA (Fortran Matrix Analysis), a library of subroutines which allows the dynamicists to solve the set of mathematical equations for given problems in a building block fashion. FORMA has been used successfully for the broad spectrum of loading conditions encountered for a number of different aerospace structures such as Titan, Skylab, Viking and Space Shuttle. The building block approach of FORMA uses FORTRAN call statements to subroutines of the library. Data is transferred to and from the subroutines by means of arguments to facilitate ease of interface. The programming language used is FORTRAN IV. Approximately 150 of the subroutines are written using a dense technique where all elements of a matrix are used. The matrix size is, thus, limited by the computer core size. This technique can efficiently give solutions of small and medium size problems (up to approximately 150 degrees of freedom). The remainder of the subroutines in the library are written using a sparse technique where only non-zero elements of a matrix are used. The matrix size is essentially unlimited with the sparse technique.

The FORMA library includes subroutines to read matrix input data from card or tape; output matrix data onto paper, card, tape or plots; perform matrix multiplication, inversion, eigenvalue, simultaneous equation solution; numerical integration techniques for time response calculation; and finite element mass and stiffness matrix calculation.

The FORMA method has, in general, several advantages over other methods. The major advantages are:

- o Software conversion to any computer having a FORTRAN IV compiler is simple;
- o Computer times are reasonable;
- o Analysts can set up programs for relatively complex problems with little programming experience;
- o Basic FORTRAN statements may be combined with calls to FORMA subroutines to give extreme flexibility in writing a program;
- o Subroutine additions and revisions are easily accomplished.

Subroutines to calculate mass, stiffness, load transformation and stress transformation matrices (subroutine FINEL) as well as vibration mode shapes and frequencies form an important part of the FORMA library. The finite element library in subroutine FINEL include rod, bar, triangular and quadrilateral plate elements. Finite elements for the tetrahedron solid, pentahedron solid, triangular fluid, tetrahedron fluid and pentahedron fluid exist. Element data may be automatically generated for regular shaped structures such as flat plates and cylindrical shells. The size limitation is approximately 6000 degrees of freedom. Modal subroutines have been programmed using variations of the Rayleigh-Ritz method of modal analysis. The system has been shown to be effective (in terms of computer time and accuracy) for the calculation of vibration mode shapes and frequencies. Computer programs (subroutines) employing this method and using both composite structure and substructure techniques have been developed. The composite structure technique employs two separate programs written in banded and sparse programming logic. Dense programming logic was used for the substructure technique. The composite structure program has been exercised on a problem with 4000 degrees of freedom. Generally, computer times compare very favorably with other programs.

With these analytical programs available, effort is currently underway to investigate ways of minimizing the cost of loads cycle analyses, which currently require coupling the analytical models of the booster vehicle and payload, and generally require 3 load cycle analyses - preliminary, design and verification. A current study[1] being performed for the Langley Research Center, is developing analytical techniques to determine the dynamic interaction between the launch vehicle and any payload, having evaluated the booster responses

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1. Contract NAS1-14370, Study of Advanced Techniques for Dynamic Flight Data Interpretation and Application for Shuttle Payloads

only once for a given event such as main engine shutdown. The analytical techniques being investigated involve developing the capability of removing the dynamic effects of payload A from the booster/payload interface, so that the response and loads of payload B can be determined without having to rerun the complete booster/payload analytical model. The analysis assumes that the external forcing function is the same for each launch vehicle and payload combination. A check on the validity and accuracy of the technique will be made using flight data from Titan/Centaur missions.

The technique, if proved successful, will be particularly applicable to the AMPS and Labcraft type payloads where science experiments will change from mission to mission. The responses and loads for new payload components and support structure could be determined from knowledge of earlier flight integrated pallet systems.

The dynamic loads for an AMPS type payload will be determined analytically using the tools available in the FORMA library. The analytical model will be verified by utilizing results of a modal survey test conducted on an integrated pallet.

Vibroacoustics - The current predicted acoustic environment for the orbiter payload bay will produce random vibration environments for payload components and subsystems which potentially could result in severe cost penalties in testing and in failures during both test and flight. Consequently, a number of studies have been performed to alleviate this problem. One of these studies (reference 2) investigated the use of payload shrouds and the potential impact/benefit for shuttle payloads. A conceptual overall shroud configuration for AMPS type payloads is shown in Figure 5.2.3-1. With the module type shroud concept, protection could be provided for 1, 2, or 3 pallets as required.

A cost model was developed to estimate potential cost savings for shuttle payloads, as follows:

$$\frac{\Delta c}{C_o} = \frac{N_P\ C_M}{C_o}\ (1 - k) - e^{0.122}\ (\frac{1}{K} - 1)$$

where: Δc = cost savings

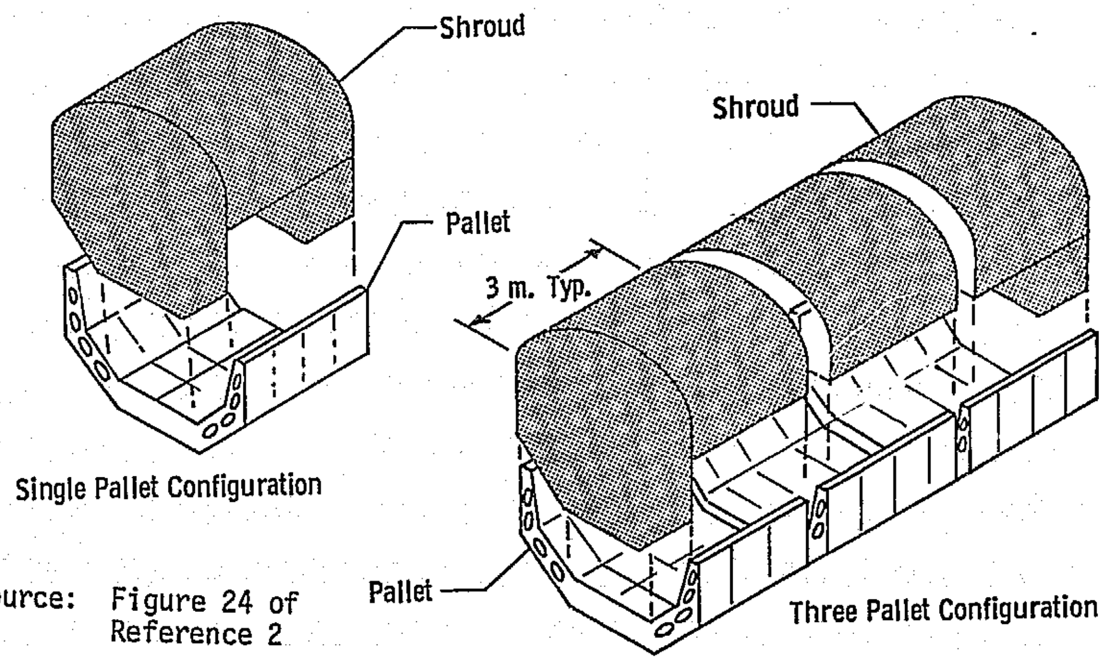
C_o = fixed cost of shroud

N_P = number of component failures expected at unsuppressed levels

C_M = average cost of modifying a component

k = noise suppression factor

Potential cost savings have been estimated for two values of noise suppression, 5 dB ($k = 0.562$) and 10 dB ($k = 0.316$) for "simple" and "complex" payloads. A "simple" payload is defined as being made



Source: Figure 24 of Reference 2

Figure 5.2.3-1 Modular Shroud Configurations For Pallet Mounted Payloads

up of 20 components/subsystems of which 10 would be expected to fail during the vibration test program based on failure rate data obtained during the study. A "complex" payload is one made up of 100 components, and the fixed cost of the shroud, including design, development and testing was estimated at $1,000,000. The cost savings as a function of number of payload test programs is shown in Figure 5.2.3-2. Note that under the assumed conditions, a loss will be incurred until a minimum number of test programs are completed; 4 to 5 for "complex" payloads, and 20-30 for "simple" payloads. The cost model does not include the launch cost (cost per pound) of flying the shrouds, or the savings resulting from decreased flight failures. However, extending the results to the total shuttle program indicated potential cost savings of 26.5 million dollars. From the vibroacoustic environment viewpoint, the use of shrouds appears highly desirable; however, other considerations such as thermal, contamination, deployment, stowage, and interface problems must be addressed.

The initial random vibration criteria for AMPS experiments and components will be established using data banks developed under the Saturn, Titan, Skylab and Viking programs. In this approach, measured vibration spectra from similar structure will be adjusted for differences in acoustic levels and structural/component weights and

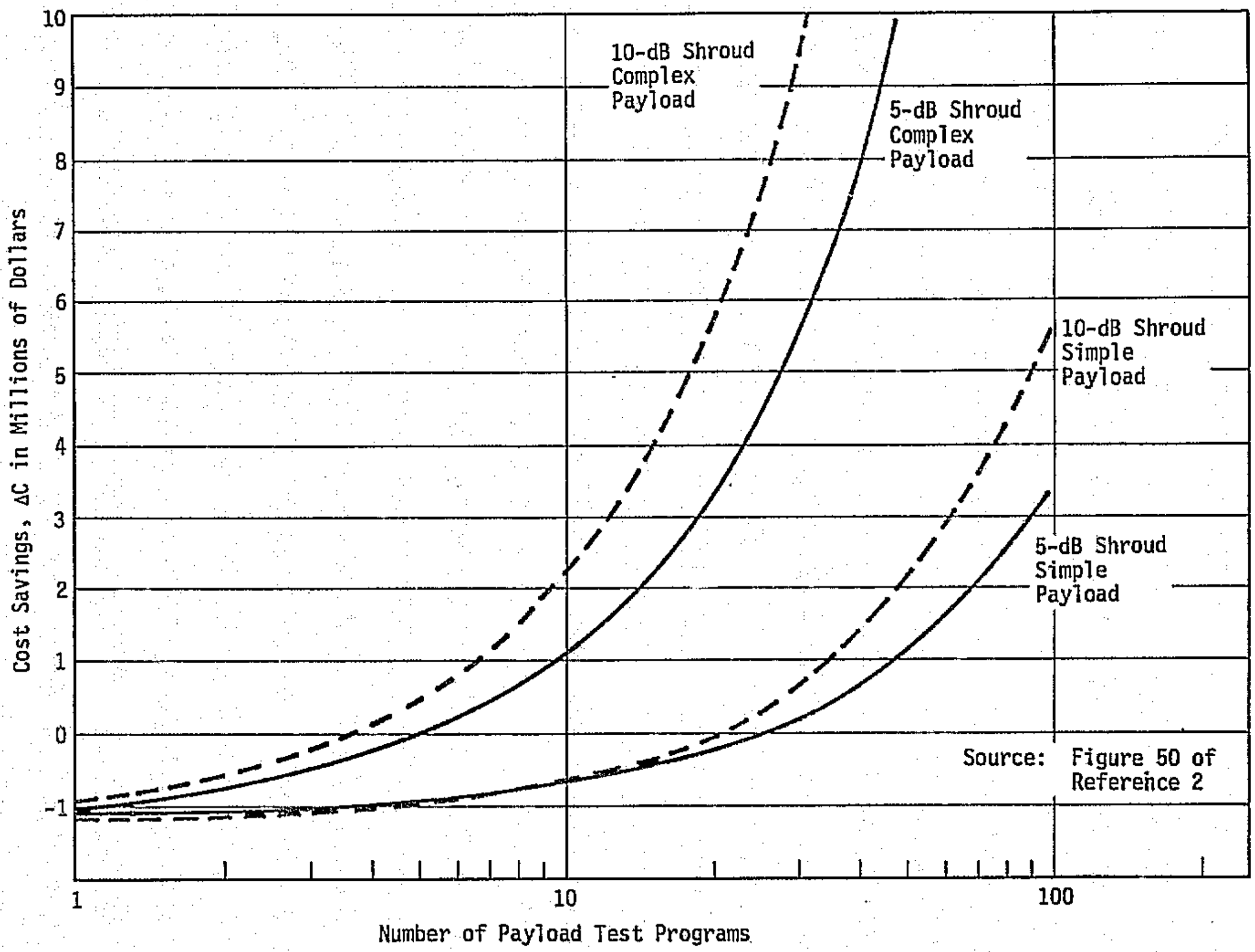


Figure 5.2.3-2 Cost Savings as a Function of Number of Payload Test Programs For Different Values of Noise Suppression

mounting configurations to establish a maximum expected flight environment. These predicted levels will be increased by a factor of 1.5 (2.25 times the P.S.D. levels) to establish the qualification test criteria for components and subsystems, in accordance with GSFC practices described in reference 4.

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1. C. V. Stahle and H. R. Gongloff, Vibroacoustic Test Plan Evaluation, Volumes 1, 2 and 3, G. E. Document, 76 SDS4223, General Electric, Space Division, Valley Forge Space Center, Philadelphia, Pennsylvania, June, 1976 (Contract NAS5-20906).

2. W. P. Rader, et al, Analytical Trade Study of the STS Payloads Environment, Report MCR-76-166, Martin Marietta Corporation, Denver, Colorado, March, 1976 (Contract NAS8-31535).

3. W. P. Rader, Stanley Barrett, and K. R. Payne, A Study to Define an Inflight Dynamic Measurement and Data Applications Program for Space Shuttle Payloads, Report NASA CR-144892, Martin Marietta Corporation, Denver, Colorado, November 1975 (Contract NAS1-13377).

4. Anon., General Environmental Test Specification for Spacecraft and Components, S-320-G-1, Goddard Space Flight Center, Greenbelt, Maryland, reprinted May 1972.

5.2.4 Structural Test Philosophy

Qualification Testing - The government and the payload community will both benefit if a common basis and qualification test factor can be used for the establishment of structural test criteria such that maximum benefit can be gained from the use of common/standardized components for shuttle payloads. Because of the differences in test philosophy in current use throughout the industry, an experiment previously flown and qualified for one government agency would not necessarily be considered qualified by a different agency. The effects of these differences in qualification test factors and test durations on test program costs were examined (reference 1) and are shown in Figure 5.2.4-1, the basis of which is the equation

$$\frac{\delta_c}{C_M} = \lambda_G \, T \, N_T \, (\gamma - 1)$$

where: δ_c = the increase in cost associated with testing at a level γ_G compared to testing at level G

γ = the test factor, $\geq$ 1.0

G= average constant failure rate associated with testing at level G

T = test duration

N_T = total number of components

C_M = average cost of modifying and replacing a failed component

Based on a nominal qualification test level G of 1 g rms, and a failure rate derived from Saturn data, the above equation simplifies to:

$$\frac{\delta_c}{N_T C_M} = 0.003 \, T \, (\gamma - 1),$$

which is plotted in the figure for various test durations and qualification test factors used by various companies and agencies. It is significant to note that the difference in the lowest and highest test factors used represents a potential factor of 4 in the cost penalty ratio. In addition, the data indicates that if there is to be no increase in cost penalty associated with increasing test duration, the test factor should decrease with increased exposure time. This approach would tend to alleviate the problem of overly conservative stresses applied to a multiple mission experiment which would occur as the result of the nonlinearity of the stress-life (S/N) curve if the same factor used for a single mission were applied. The problem of exposure times becomes particularly relevant to system level acoustic tests where a payload may be made up of both single and multiple mission experiments. This entire area of test factor and duration margins,

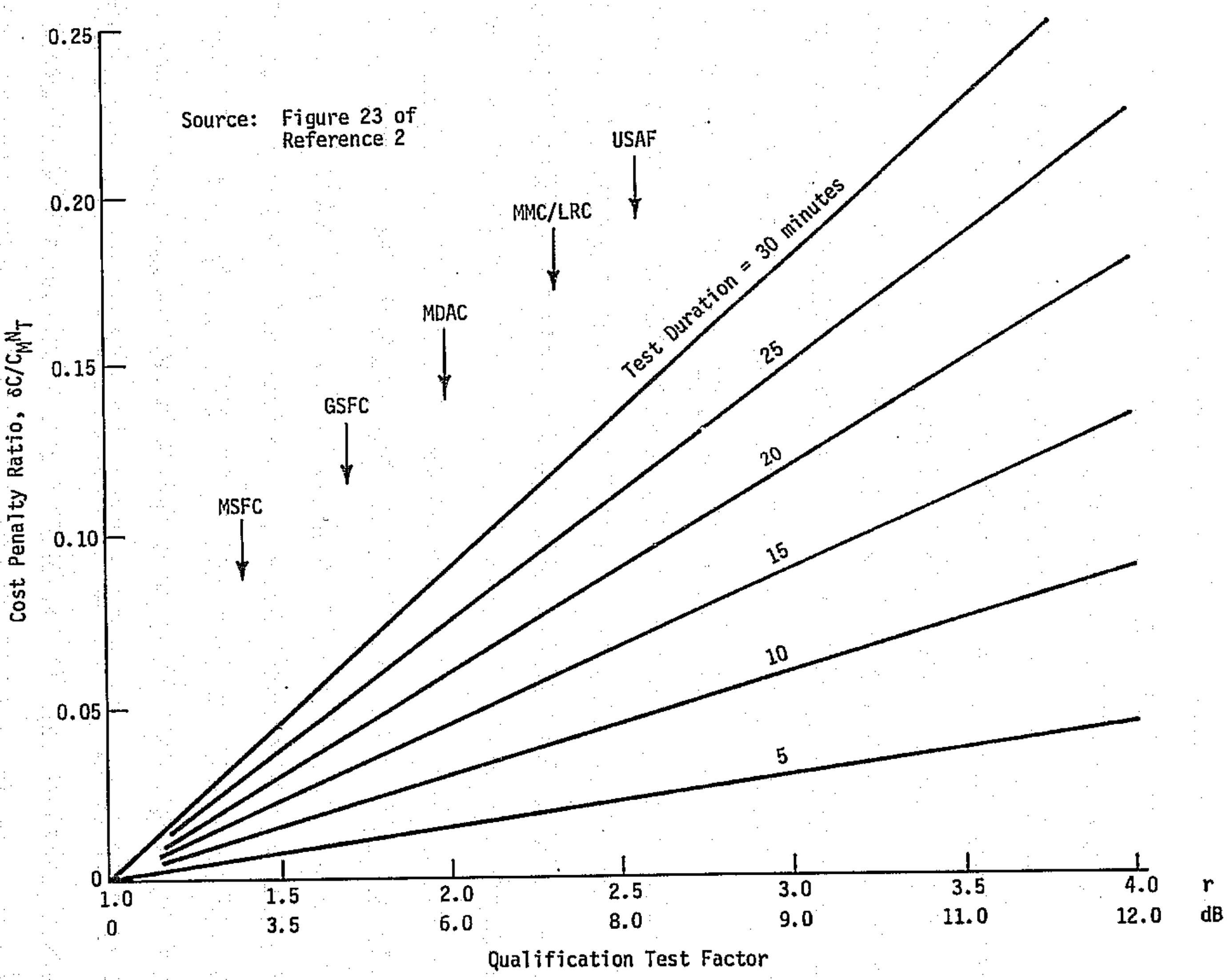


Figure 5.2.4-1 Effect of Qualification Test Level Philosophy on Test Costs

and level of assembly testing requires additional study and coordination throughout the industry if a cost effective philosophy is to be realized for the Shuttle program.

<u>Modal Survey Test</u> - In the low frequency region controlled by vehicle dynamics, modal survey testing will be used to verify the analytical model of the AMPS payload. Member loads and responses will be calculated for orbiter forcing functions as discussed previously in Section 5.2.3, Vehicle Dynamics.

For this test, either flight type experiments and components or dynamic simulators can be used. The interface fittings between the pallet and the payload bay should be duplicated to provide the proper degrees of freedom at those points. With a test verified analytical model of mode shapes, frequencies, and damping, for the AMPS payload coupled with the orbiter model, dynamic loads in structural members and motion at experiment mounting points can be determined for the vehicle transient events and on-orbit forcing functions. The modal survey test will need to be conducted only once unless major configuration changes are made to the AMPS payload.

Sinusoidal testing should be performed only for those experiments and subsystems for which analyses indicate the design may be marginal, or where alignment or pointing accuracy must be verified by test.

<u>Vibro-Acoustic Testing</u> - The predictions for acoustic levels in the payload bay are higher than those of current launch vehicles, and, unless reduced, can result in severe cost penalties for shuttle payloads, particularly if current test philosophy and qualification factors for the establishment of test criteria are continued to be used. Compounding the problem are such factors as:

(1) Reliability requirements for single mission spacecraft and experiments as compared to multi-mission, repairable spacecraft;

(2) Commonality/standardization of components and subsystems, and qualification of components by similarity from previous flight usage; and

(3) Cost effectiveness of the protoflight versus the prototype system level tests for different types of payloads and mission requirements.

In the vibroacoustic frequency region, it is assumed that current requirements and test philosophy will be applied in the test programs for initial payloads, until adequate test and flight data are acquired to establish statistical confidence in the environment with the potential of reducing test factors. For the initial payloads, qualification testing should be conducted on a component and subsystem basis using random vibration or acoustic excitation depending on the level

of assembly and susceptibility to acoustic loading. Assembly level testing of integrated pallets should be conducted at the maximum expected flight environments as discussed in Section 5.2.3, Vibroacoustics.

Each of the 3 integrated pallets for Flight 1 should be subjected to an acoustic test to the maximum expected flight environment to verify the structural integrity of the equipment mounting bracketry and obtain data to verify/modify the random vibration criteria. Testing of individual pallets will be adequate since there should be no significant interaction effects between pallets due to acoustic loading. Data obtained from these tests and vibration and acoustic data from early shuttle flights will form a firm basis for establishing realistic vibration criteria for testing future experiments at the component/subsystem level. Further system level testing will not be required unless major changes are made in the payload configuration.

Static Testing - As the structural design safety factor is increased, less testing is required to assure structural integrity. If FS = 1.4 is used, a static test article is required and would be subjected to an ultimate load test. If FS = 2.0 is used, no separate test article is required, but the flight article would be subjected to a proof test. If FS = 3.0 is used, no static testing is required.

As the structural factor of safety is increased, the additional material required typically produces a small cost increase compared to the cost of a test article and/or static test program.

AMPS/Labcraft type support structures, for the flights evaluated, amount to only 6% to 7% of the pallet mounted payload weight.

Since the resulting payload weights are well within the Orbiter/Spacelab performance capability, a higher priority will be assigned to cost rather than weight. It follows that the most desirable candidate approach is the FS = 3.0 structure with no static test article or proof test requirement.

Qualification and Acceptance Test Plans - Options and Recommendations - For the first mission, the conventional, prototype philosophy test plan, shown in Figure 5.2.4-2, consists of a qualification test program and acceptance testing of flight components. The equipment/experiments are comprised of both "new" and "old" components; i.e., components which have been previously qualified on other programs. The qualification criteria for these "old" components must be evaluated with regard to the shuttle payload bay criteria. The test program outlined in the figure is designed to produce maximum reliability with associated maximum costs. To reduce costs, with an associated increase in risk, the options exist to eliminate one or more levels of testing, designated A, B and C corresponding to components, subsystem, and integrated pallet levels of assembly respectively. Additional options exist in the treatment of a failure at the various levels of assembly. Depending upon the failure evaluation, the failed

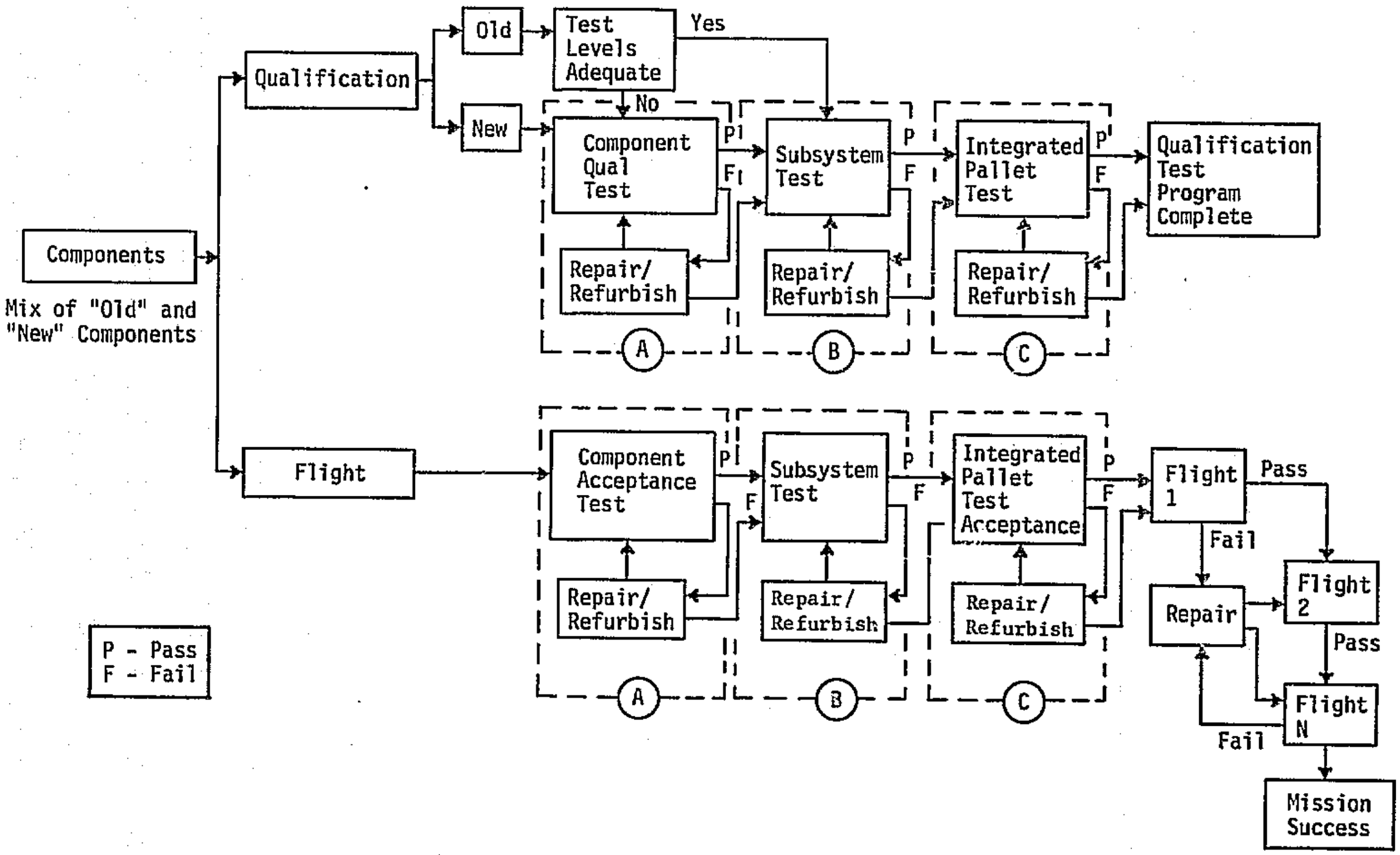


Figure 5.2.4-2 First Flight Prototype Test Program

component may be repaired and retested at the same level of assembly, or advanced to the next level of assembly test.

An alternate test program, the protoflight concept, is depicted in Figure 5.2.4-3. In this approach, the flight hardware would be tested to qualification levels for acceptance durations. As with the prototype program, options exist to eliminate one or more level of assembly tests (A, B, or C). with an associated increase of risk of flight failure. The decision to eliminate one or more levels of testing must be based on evaluation of the mission requirements, consequences of flight failure or loss of data and repair/refly capability.

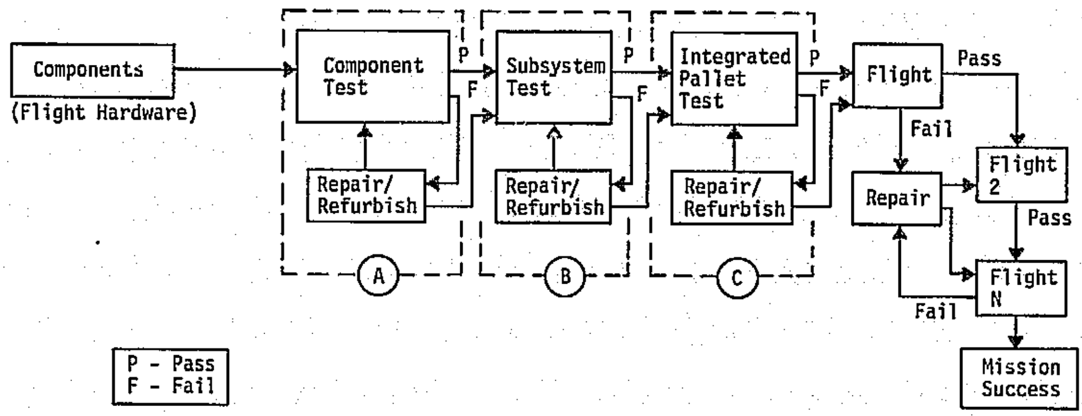


Figure 5.2.4-3 First Flight Protoflight Test Program

For the second and subsequent missions, the test program for new instruments/equipment could be modified as shown in Figure 5.2.4-4, in which both the prototype and protoflight concepts are shown. In this case, it is felt that the integrated pallet test could be eliminated in the qualification test program, with the option of eliminating the component or subsystem level testing, again depending on the mission requirements and degree of risk acceptance. For the flight components, it is felt that the subsystem level test is adequate for either the prototype (acceptance) or protoflight approach, based on the premise that sufficient data will be acquired during the previous integrated pallet tests and flight to define realistic criteria for these tests.

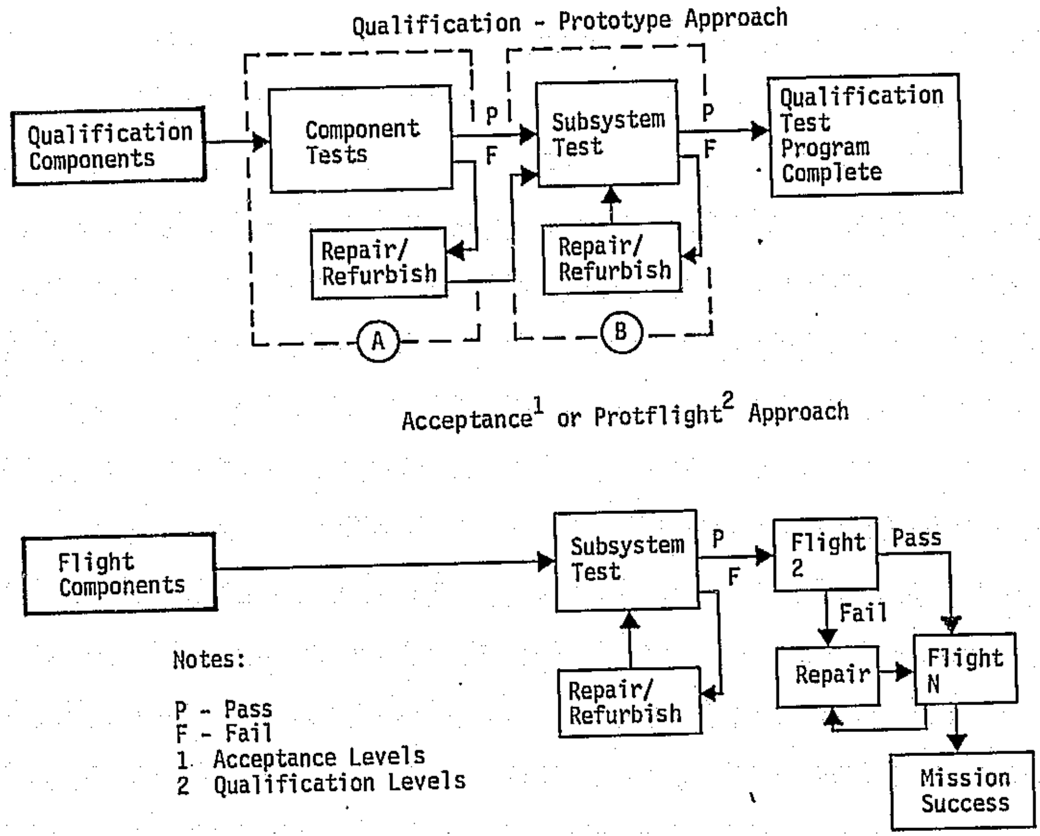


Figure 5.2.4-4 Alternate Test Plans for Flight 2 and Subsequent

A recommended approach for AMPS payloads is summarized below.

(1) <u>Flight 1</u>

<u>Qualification</u> Experiments/Components Requiring 3 or More Sets of Hardware	Component Level Tests (Vibration)	Integrated Pallet Tests (Acoustic)
<u>Acceptance or Protoflight</u> Experiments/Components Requiring 1 or 2 Sets of Hardware	Component Level Tests (Vibration)	Integrated Pallet Tests (Acoustic) Only for Protoflight Approach

(2) Flight 2

Qualification

Experiments/Components Requiring 3 or More Sets of Hardware	Subsystem Level Tests

Acceptance or Protoflight

Experiments/Components Requiring 1 or 2 Sets of Hardware	Subsystem Level Tests

REFERENCES

1. W. P. Rader, et al, Analytical Trade Study of the STS Payload Environment, Report MCR-76-166, Martin Marietta Corporation, Denver, Colorado, March, 1976 (Contract NAS8-31535).

5.2.5 Payload Substructure Study

A 22½ day maximum pallet turnaround time at the Level IV integration site is being considered. For a complex pallet payload, off-pallet build up and test must be investigated as a way to provide more integration and checkout time. This discussion presents the structural aspects of a separate pallet payload substructure. This slip-in structure or liner would allow payload build up including structural attachments, plumbing, cabling, cold plates, and flight support equipment. A listing of design requirements follows along with a discussion of two design alternates.

5.2.5.1 Requirements

A basic list of requirements as derived from general design criteria and the AMPS layouts is presented below.

(1) Assume payload is pallet modularized (i.e. no instrument structural tie or attachment across pallets).

(2) Provide for complete payload build up and checkout.

(3) Provide access to existing pallet hardpoints for instrument or substructure attachments.

(4) Provide access to pallet subsystem equipment and interfaces.

(5) Assume pallet substructure will use a strongback type structure for support during ground use.

(6) Structure will have matched or coordinated hole patterns to interface with the pallet hardpoints.

(7) Structure will provide the same load carrying capability as the pallet.

(8) Assume that pallet simulators (GSE) are available for payload build up and transport.

5.2.5.2 Discussion

Two designs were evaluated as possible payload substructures. The two concepts were a pallet within a pallet structure and an intermediate structure with direct payload to pallet hardpoint mounting. The pallet within a pallet approach uses a complete stable structure similar to an equipment truss. This structure is attached to the pallet hardpoints and provides separate mounting points for the equipment and instruments. A conceptual sketch of a possible configuration is shown on Figure 5.2.5-1. This sketch shows the general concept features and was not configured to match any of the AMPS pallet layouts. An independent structure would be designed for each pallet to match instrument attachments and to satisfy other considerations such as field of view,

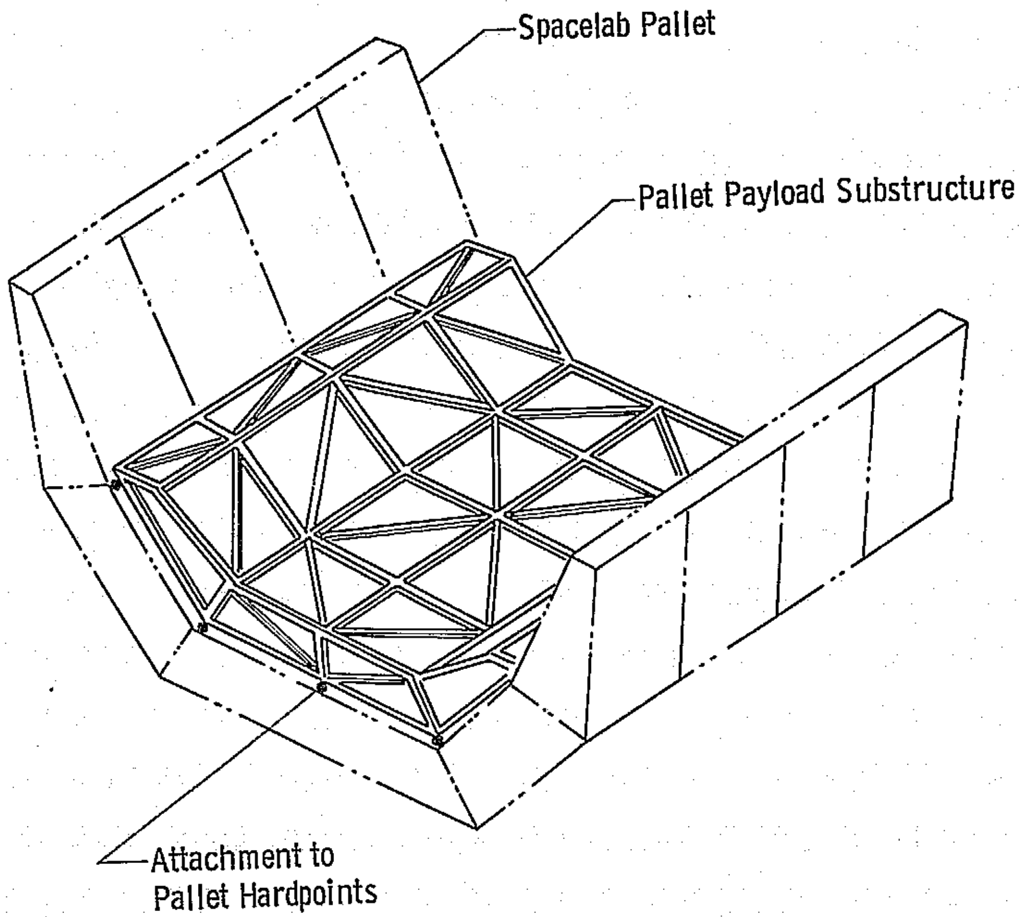


Figure 5.2.5-1 Pallet Within a Pallet Concept

operational interfaces, etc. Advantages of this design are: strength, stiffness, the ability to match the structure to the instruments, and ability to build up and transfer a complete payload to the pallet. Disadvantages are: loss of multiuse capability, weight, loss of usable volume, stiffness interaction between pallet and structure, and increased costs due to complex structural modeling and tests. The intermediate structure approach or pallet liner is shown on Figure 5.2.5-2. Here the liner is non-structural in terms of supporting the instruments and major support equipment. The concept allows plumbing and cabling to be laid out and attached to the liner. Instruments are attached to the pallet hardpoints which are accessible through the liner. The liner itself is attached to various instrument fittings by secondary

attachments. Advantages of this approach include: low weight, minimal loss of payload volume, and a common design that would work for many payloads. Disadvantages are: requires a complex handling sling to pick up all the instruments, requires secondary attachment between instruments and liner, and compounds the integration/interface task by adding handling interface requirements.

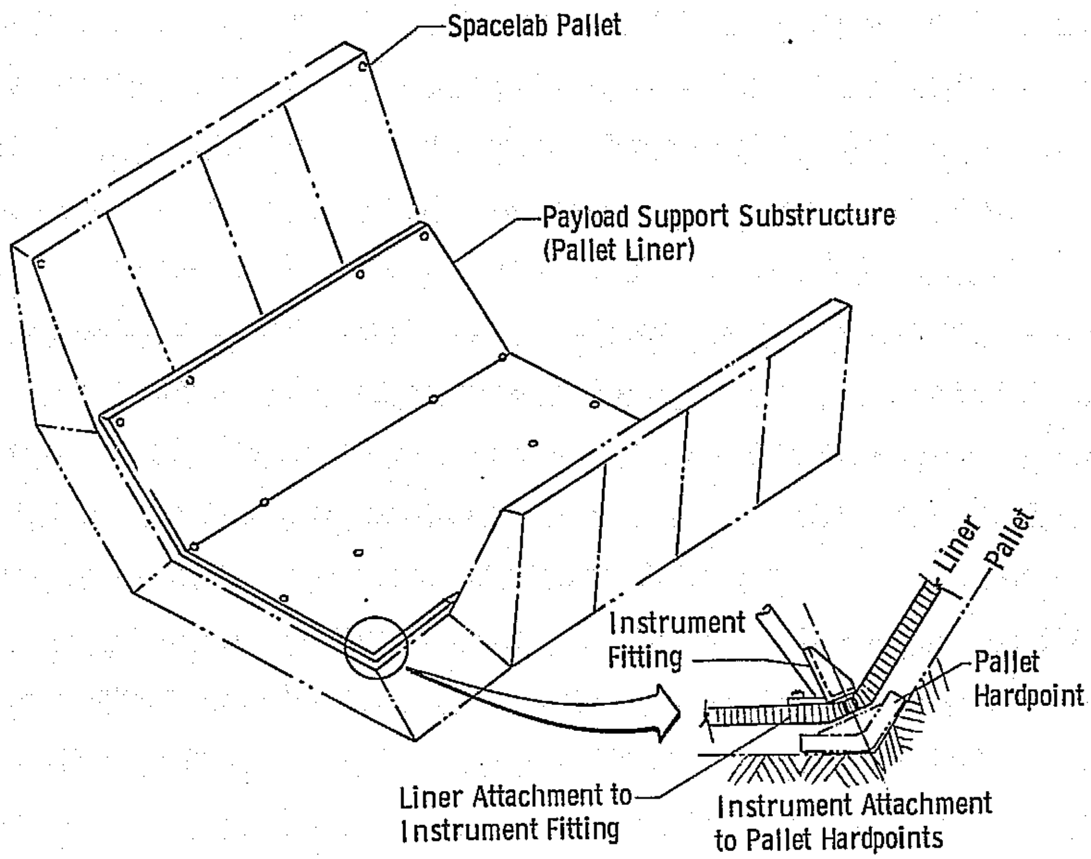


Figure 5.2.5-2 Intermediate Structure Concept

The AMPS Flight 1 layout was investigated to compare the payload support substructure concepts with the actual payload configuration. A pallet by pallet check of the instrument mountings showed that the forward and center pallets do not require a separate pallet structure. The degree of complexity involved in payload build up does not warrant the additional structure. The forward pallet has three separate items that interface with the pallet: the Miniaturized Pointing Mount (MPM) intermediate structure, the ESP module support structure and the Gas Release module support platform. Separation of the three instruments (no shared hardpoints) and noncomplex interfaces (no difficult structural attachments and no alignment requirements) make it possible to achieve the pallet turnaround time cycle. The design of the forward pallet interconnecting harness utilizes the gas release support platform as the primary routing path. Connections for adapting the MPM

and ESP harnesses as well as the standard pallet power and data harnesses would be from the platform. The thermal fluid loop is not used on this pallet so no interconnections are required.

SIPS, the Beam Diagnostics Package, and the RF terminal are located on the center pallet. SIPS will have either flown or have been pallet fit checked, so no integration problems are expected. The RF terminal consists of an antenna package and an electronics package. The electronics package is installed on a Spacelab cold plate in the standard location while the antenna is panel mounted near the sill. Neither package poses an integration problem. The Beam Diagnostics Package mounting structures are simple enough to expect that installation and checkout could be accomplished in the allotted time. Pallet harnessing design would be similar to that for the forward pallet except that the primary harness would be located off SIPS. Utilization of the standard cold plate location for mounting of the RF terminal components eliminates the need for any fluid loop plumbing.

The aft pallet points out some of the integration problems involved with a complex and densely packaged payload. An Electron Accelerator, OBIPS on the MPM, the LIDAR receiver, two LIDAR transmitters, the IECM, the Pulse Power Supply and Solar Flux Monitor are all housed on the aft pallet. Complex instrument to instrument and shared interfaces exist along with potential access problems. A dedicated Heat Exchanger, a Peaking Battery on a standard cold plate, and cold plates mounted on the instruments further complicate the integration task on this pallet. Support structure used includes direct mounting brackets, intermediate truss structures and special struts from the LIDAR attached to the pallet/orbiter fittings. All these factors contribute to making the aft pallet the most likely candidate for the payload support substructure.

A cursory examination of the pallet within a pallet concept showed it would not work on the aft pallet. Payload elements would have to be repackaged to fit within the payload envelope. Access to hardpoints below the structure would also be difficult. An investigation of the aft pallet using the liner concept also proved unpromising. There is a greater possibility of physically packaging the payload within the envelope with this approach. The added interface complexity and instrument handling requirements make this an unattractive scheme. To transfer the payload by simultaneously lifting all the instruments and the liner by use of a special sling is not good handling practice. Unsure load distribution and 2 g hoisting loads on the instruments eliminate this approach from further consideration.

With both substructure concepts proving unworkable, the last approach considered was the examination of the existing layout to see how severe the integration problem really was. The aft pallet layout, when examined in more detail, was found to have some features of the payload support substructure already incorporated. Three of these features were: the ability to build up the payload in a sizable piece capable of transfer to the pallet, the ability to verify interfaces

before the actual pallet interfaces were on hand, and the ability to build up plumbing and wiring harnesses in large segments.

The Pulse Power Supply is located on the floor of the pallet and is attached via brackets to 8 hard points on the floor. The Electron Accelerator and LIDAR Receiver are located on top of the Pulse Power Supply to reduce power cable lengths; and the Electron Accelerator is mounted directly to the Pulse Power Supply. The LIDAR Receiver has three support struts from each side. Two struts are tied to the Pulse Power Supply brackets (hardpoints) and the third strut attaches to the pallet/orbiter interface fitting on the sill. These three instruments constitute a modular segment that can be built up and handled as a unit. The OBIPS, IECM, and LIDAR transmitters are located to the sides of the Pulse Power Supply. Because of the limited hardpoints available, these instruments end up sharing six of the Pulse Power Supply hardpoint locations, and in addition, each instrument uses two separate hardpoints on the sides. Because of space limitations, the support struts from these instruments to the shared hardpoints actually interface with the Pulse Power Supply brackets. This allows an interface and fit check between these instruments and the Pulse Power Supply by providing only a GSE fixture simulating the pallet sides with six hardpoints. The other aspect of the existing layout is the ability to build up utility lines. Because of the packaging density, most of the plumbing and wiring are routed upon the instruments and support structure, and can be transferred to the flight pallet as significant interconnecting segments.

5.2.5.3 Conclusion

This analysis shows that the forward and center pallet payloads on the AMPS flight do not require a substructure to reduce pallet turnaround time. The aft pallet payload, however, does present an integration and assembly challenge. Two pallet substructures were described, one approach was to use a separate slip-in structure similar to an equipment truss. The other design approach was a non-structural liner or spacer between the payload and pallet. Evaluation results revealed that neither concept satisfactorily met all the requirements. Both concepts were eliminated for the following reasons: repackaging of the payload was required due to loss of volume, access to pallet hardpoints was restricted, and added handling interfaces (loads and physical attachments) would overly complicate the instrument design. The proposed design layout for the aft pallet does provide most of features required to minimize pallet turnaround time. These features inherent in the preliminary design are; instrument groupings that allow build up and transfer as a unit, interface verification prior to pallet availability, and utility line build up on instruments or support structure rather than on the pallet.

5.3 Thermal Control Subsystem

5.3.1 Thermal Model Description

The AMPS Flight 1 thermal math model illustrated in Figure 5.3-7 consists of 108 nodes. The model represents the major AMPS pallet thermal control elements which include the external configuration, the active liquid loop, the mini-mount thermal canister, the internal configuration, and cold biased components.

The model has been constructed for use with the Martin Interactive Thermal Analysis System (MITAS) program. Radiation thermal couplings and environmental heat fluxes have been calculated using the Thermal Radiation Analysis System (TRASYS) program. TRASYS considers solar, albedo and earth infrared heat fluxes; shadowing and reflections between surfaces are also included.

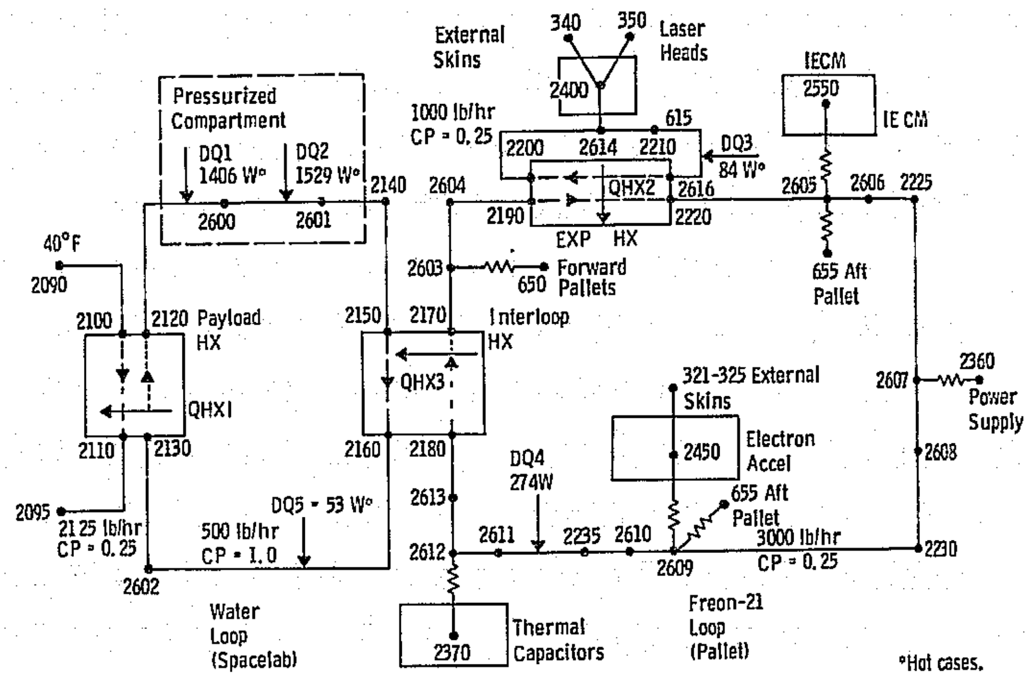
Mini-Mount Canister - The AMPS Flight 1 configuration uses a thermal canister developed as a part of our Multi-Use Mission Support Equipment (MMSE) contract for MSFC. The canister is 1 M x 1 M x 3 M and uses the Skylab Apollo Telescope Mount (ATM) liquid coolant loop concept and hardware. The canister inner walls are maintained at a constant temperature using a radiator by-pass loop, temperature sensor and a thermal control valve. The external surfaces of the canister are radiator panels. The canister liquid thermal control loop is modeled using one-way thermal conductors to represent the radiator panels coolant flow. The inlet temperature to the radiator has been assumed to be 52^{o}F.

Active Thermal Control Loop - The fluid loop model consists of 35 nodes as shown in Figure 5.3-1. The Spacelab module heat loads and pump power is a direct heat input to the coolant. The heat transfer across each heat exchanger is calculated during problem solution, using the heat exchanger effectiveness. Fluid loop performance operations have been grouped in the AMPS thermal model as a MITAS subroutine. Thermal capacitors are represented in the model using a MITAS subroutine.

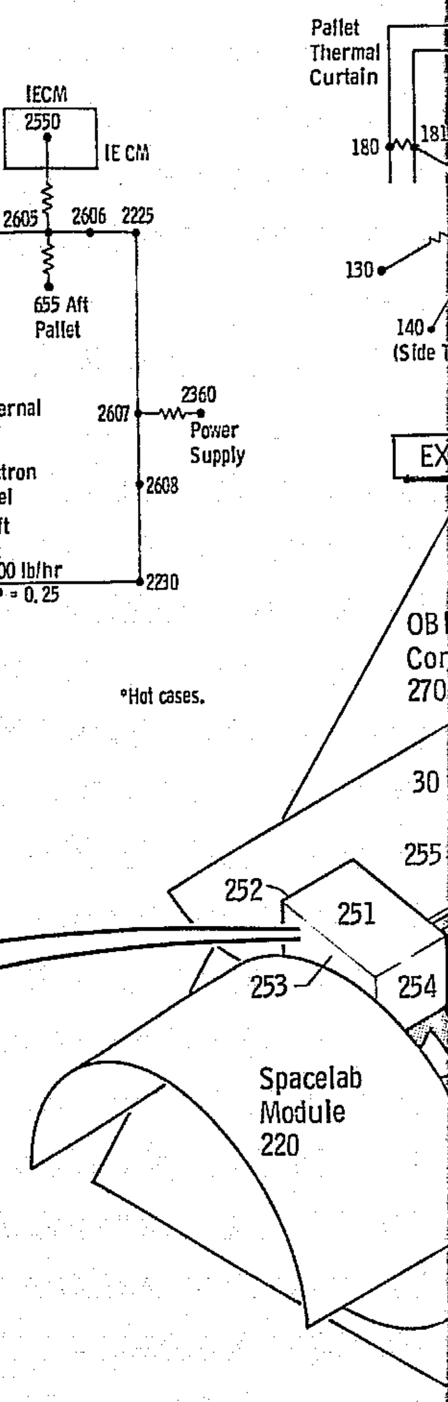
Internal Configuration - The internal configuration represents the compartment formed by the thermal curtain and the pallet. The pallets are represented by two nodes and radiation couplings are included to represent the heat leak through the curtain.

Cold-Biased Components - The thermal design approach for cold-biased components is detailed in Section 5.3.2. The cold-biased components are represented by an external and an internal node that represents the average temperature of the component. As the detailed instrument design and thermal requirements evolve with program maturity, more detailed analysis will be required.

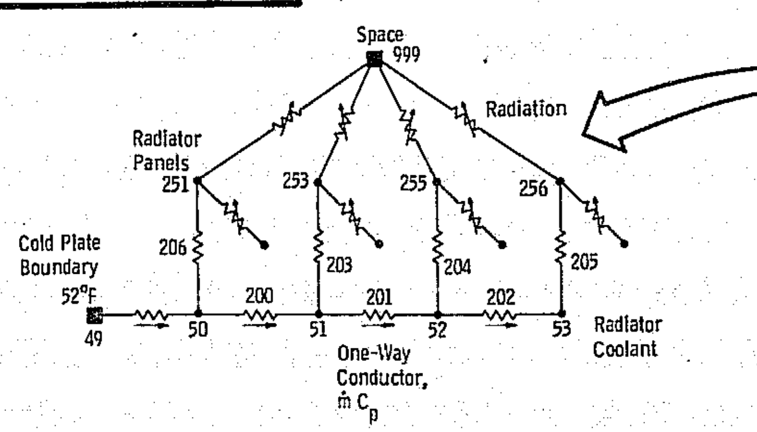
FLUID LOOP



INTERNAL CONFIGURATION



MINIMOUNT CANISTER



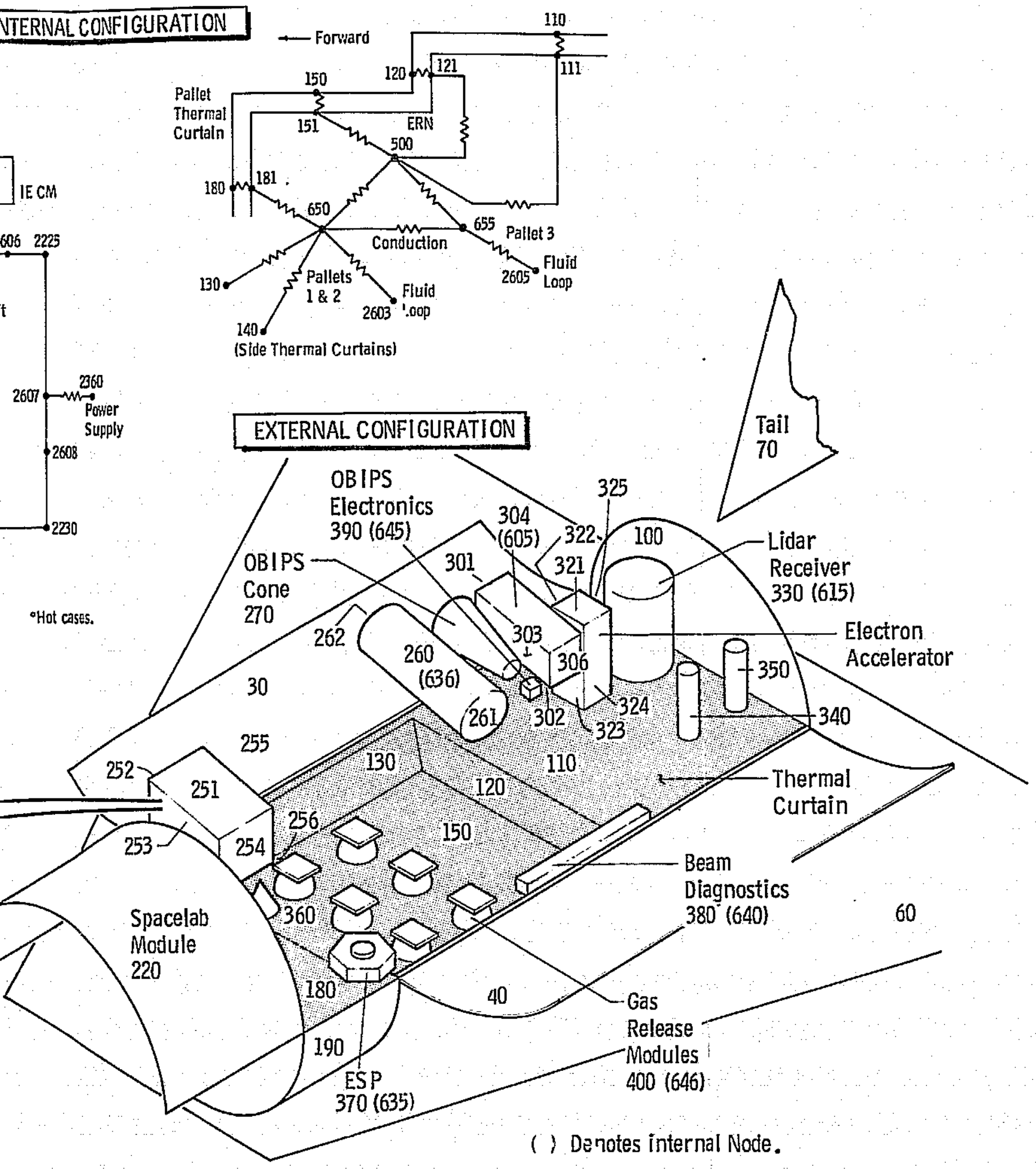


Figure 5.3-1 AMPS Flight 1 Thermal Math Model

FOLDOUT FRAME 2

It should be noted that the Flight 1 configuration does not include a SIPS thermal canister. However, a SIPS canister is planned for subsequent AMPS flights and for other payloads. To facilitate future analysis, the cryo-limb scanner has been replaced, for analysis purposes, by a SIPS sized canister. The AMPS Flight 1 model can be used to determine heat rejection characteristics for a SIPS canister in a representative location. The SIPS canister in the Flight 1 model does not affect analysis results for Flight 1 because the canister provides approximately the same radiation blockage as the cryo-limb scanner that it replaces. Additionally the IR spectrometer is included in the model and thermal performance (cryogen usage) of the cryo-limb scanner is similar to the IR spectrometer.

5.3.2 Cold-Biased Components

A cold-biased thermal design is well suited to AMPS deployable packages and several other instruments because it is simple and low cost. The instrument package is designed to be relatively cold for hot conditions and electrical heaters are provided for cold-case operation.

The thermal design approach for the environmental sensing package (ESP) is typical and is illustrated in Figure 5.3-2. All cold-biased components are partially insulated with multilayer insulation (MLI), and silver-coated teflon (SCT) radiation areas, of tailored size, are used to reject heat. SCT is a low α/ϵ surface resulting in low temperatures even for full solar conditions.

Analysis, using the AMPS Flight 1 thermal model, has been completed for the six cold-biased instrument packages to determine the average temperature as a function of percentage of MLI as shown in Figure 5.3-2 The upper temperature limit for these packages is 122°F (50°C) and the corresponding percentage of MLI is selected. Heater power is then calculated as a function of the average package temperature. Several instrument packages that are deployed from the AMPS do not operate when stowed on the pallets and require heater power for hot conditions. Mission-average heater power is determined using the Orbiter attitude requirements and the worst case heater power.

5.3.3 Thermal Capacitor

Peak temperatures occur in the AMPS pallet liquid loop during operation of the Electron Accelerator and the LIDAR. These instruments are operated non-concurrently and dissipate 5000 watts for 0.6 hours of a 1.5 hour orbit with a relatively low standby power for the remainder of the orbit. Mission timeline analysis calls for operation of the LIDAR or Electron Accelerator for periods of four orbits and large temperature fluctuations of the coolant temperatures occur. A study has been completed to determine the effect upon the AMPS fluid loop temperatures of the Spacelab thermal capacitors.

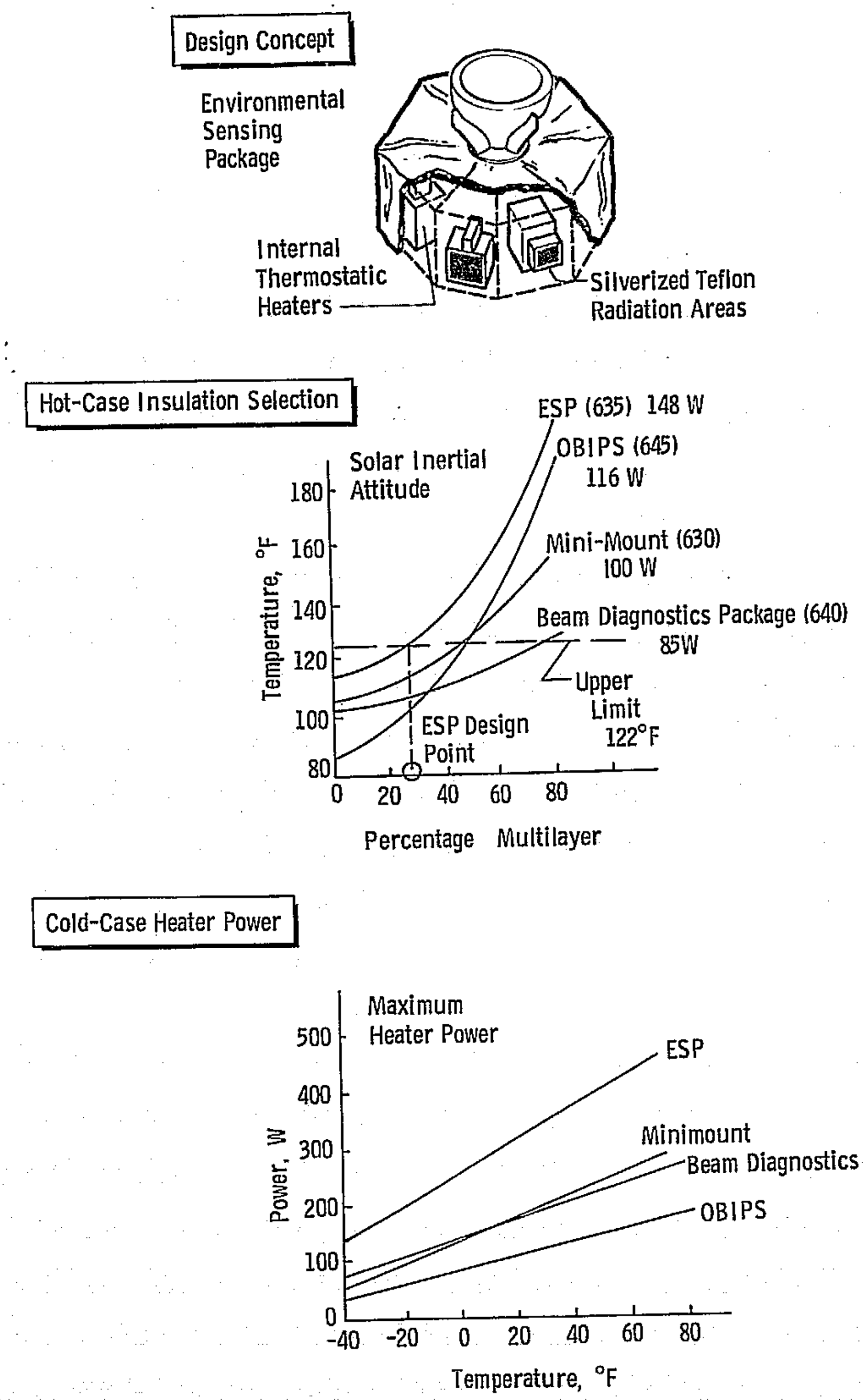
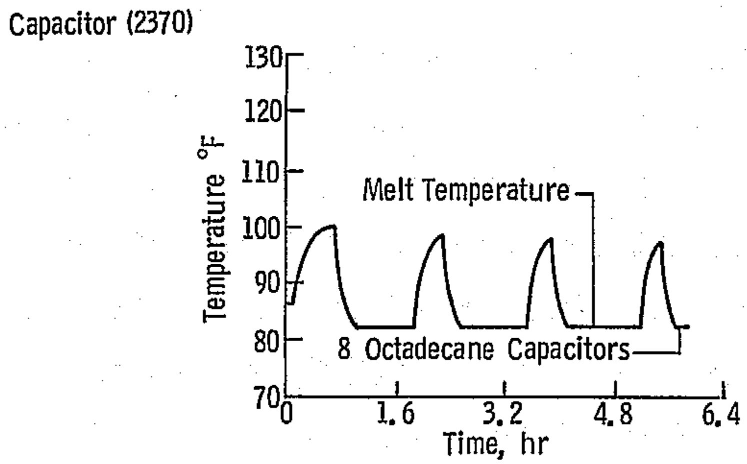


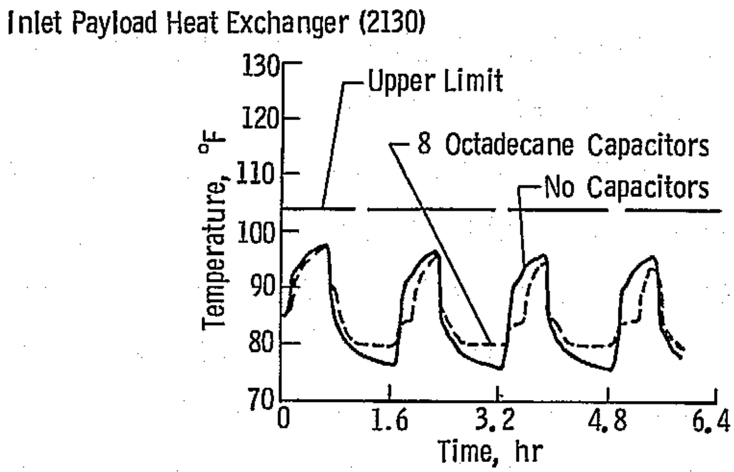
Figure 5.3-2 Cold-Biased Thermal Design Approach

Four thermal capacitors are available to the payload from Spacelab. Each capacitor is a rectangular container that in the standard configuration, mounts on one Spacelab coldplate and provides a volume of 0.23 ft^3 for storage of a wax. During phase change (freezing or melting) the wax temperature remains constant and this constant temperature serves to minimize fluctuations in the pallet loop coolant temperatures. Thermal performance of the capacitor is a strong function of the size of the capacitor(s) (heat required to change the wax from a solid to a liquid) and the thermal coupling from the wax to the fluid.

Maximum reduction in coolant temperature fluctuations is achieved when the wax phase change temperature is equal to the average coolant temperature. The Spacelab capacitors are supplied with Heneicosane wax that has a phase change temperature of 104°F (40°C). This temperature is the upper temperature limit of the pallet coolant and does not provide optimal performance because of the normally low average temperature of the coolant. Octadecane wax was selected for the present studies because the phase change temperature is 82°F that is more consistent with the average temperature of the pallet loop coolant during operation of the electron accelerator and the LIDAR.

The goal of the present studies is to determine the quantitative reduction in coolant and component temperature fluctuations that result when the thermal capacitors are used. The thermal capacitors have been located at the outlet of the Electron Accelerator, and the studies have assumed eight capacitors (Octadecane) as a practical upper limit. The analysis results are presented in Figure 5.3-3, where the temperatures of selected fluid loop locations (with and without capacitors) are presented as a function of time. These data have been obtained using the overall AMPS flight/thermal model described in Section 5.3.1. These results show that the Electron Accelerator average temperature fluctuation is reduced by less than 2°F. Similarly, the temperature fluctuation at the inlet to the payload heat exchanger is reduced by less than 4°F. Thermal capacitors have not been included in the baseline AMPS Flight 1 and 2 configurations and similar results are expected for subsequent missions.





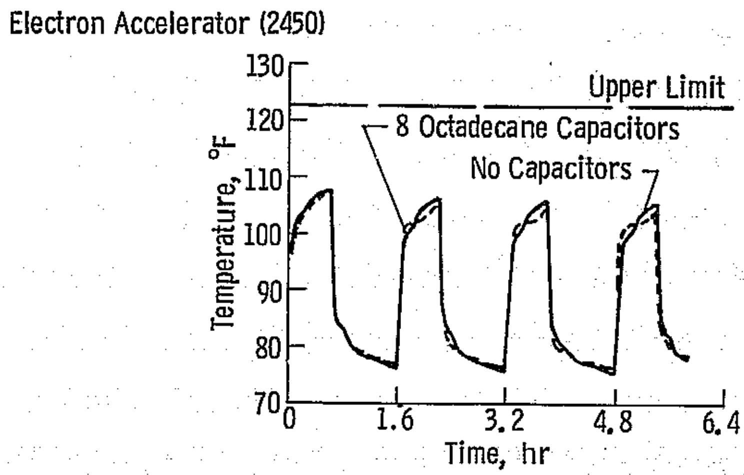


Figure 5.3-3 Thermal Capacitor Performance

5.4 Electrical Power and Distribution Subsystem

5.4.1 Hardwire vs RF Trade Study

Deployable integrated equipment modules are included in both the AMPS Flight 1 and Flight 2 configurations. A number of these modules ar removed from the pallet by the Shuttle Remote Manipulator Arm, maneuvered by the arm to various locations above the payload bay during instrument operation, and then returned to the pallet for return to earth with the Shuttle. Figure 5.4.1-1 shows a typical module on the Remote Manipulator Arm during the instrument operation.

The power, data and command interface with the integrated equipment modules must be maintained at all times during release, remote operation and return to the pallet. To maintain this interface requires either a hardwire connection to the module at all times or a self-contained power, data and command configuration within each module. A trade study was completed to determine the optimum configuration at the minimum cost to accomplish the above objective.

5.4.1.1 Configuration Definition

The objective of this trade study is the selection of a design configuration to provide electrical power, data and commands to the remote modules when the modules are separated from the pallet mount. Two basic configurations were considered: hardwire from the pallet to the module, and self-contained power, data and command configuration within each module. Figure 5.4.1-2 is a block diagram summary comparison of the two approaches.

Hardwire - Two methods have been identified to provide a hardwired link to the remote modules: cabling that routes from the pallet along the Spacelab structure to the Remote Manipulator Arm with connection to the module through a connector built into the arm end effector, or direct connection from the module to the pallet using a cable management system.

The first configuration requires that the cable be routed from the AMPS payload, along the Spacelab structure to the mechanical interconnect with the Remote Manipulator Arm and then routed along the arm to the end effector for connection to the integrated equipment module. Interconnection to the module would be provided when the end effector engages and disconnection would occur when the module is returned to the pallet. Figure 5.4.1.3 presents a schematic representation of the cable routing.

Two approaches are possible for providing a hardwire cable link to remote modules via the Remote Manipulator Arm. Both methods involve the construction of the cabling in segments that would be installed on different sections of the Spacelab between the AMPS pallet and the arm end. One approach would be for the AMPS contractor to

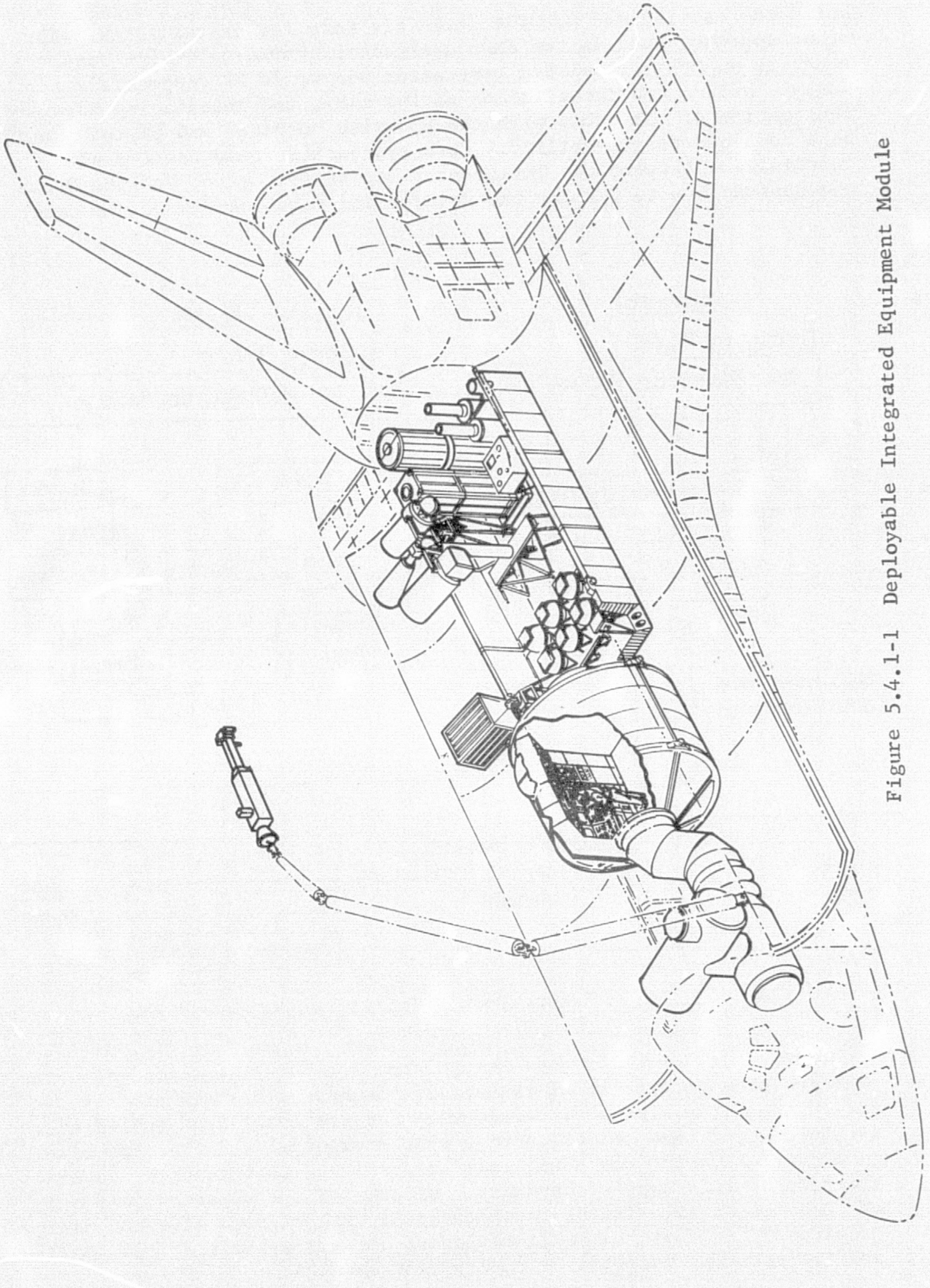


Figure 5.4.1-1 Deployable Integrated Equipment Module

build the cabling and provide it in kit form for installation. The other approach entails the AMPS contractor providing design requirements to an Orbiter related contractor who would be responsible for design, build, and installation of the cable and interfacing hardware. Integration of the cable with the Spacelab, Orbiter and payload would be a combined responsibility. Both approaches would require close coordination between the Intercenter Interface Control Working Group and contractors to resolve the details and requirements of the design.

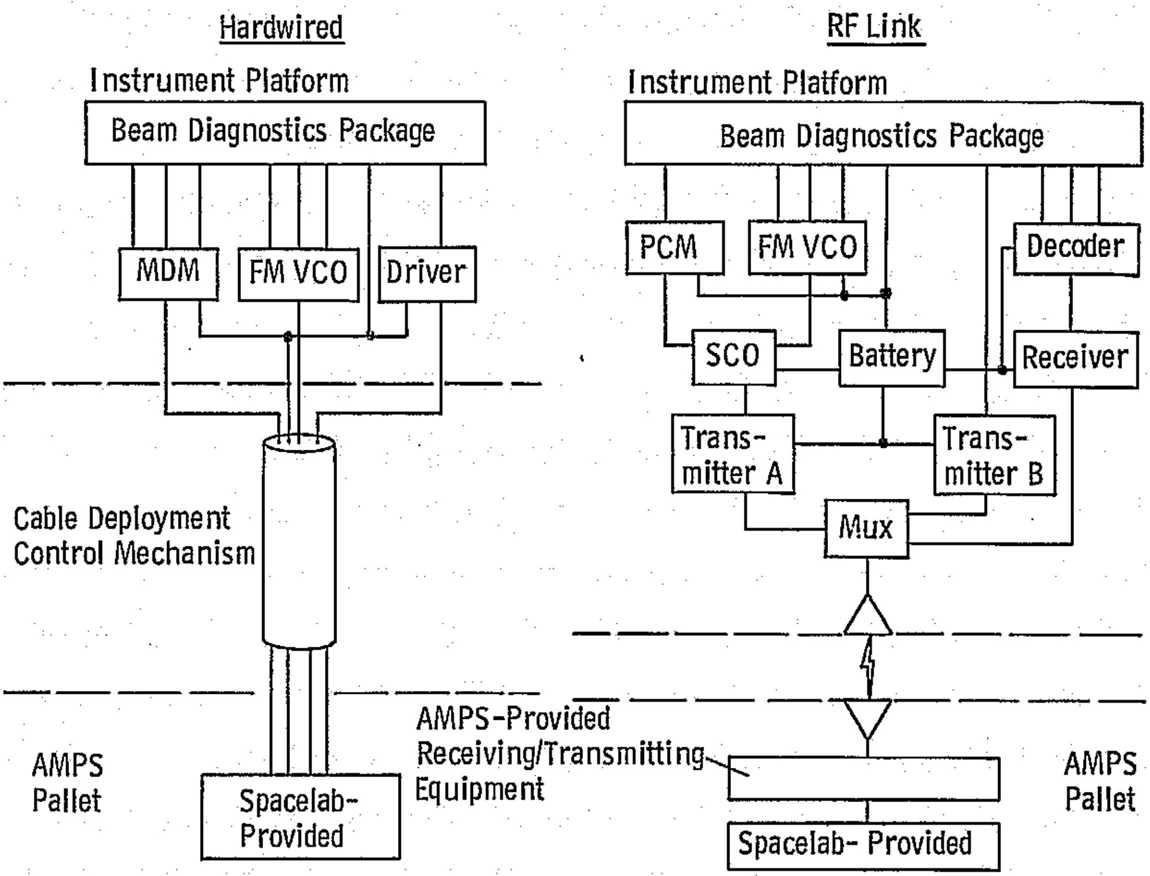


Figure 5.4.1-2 Remote Module Configuration Comparison

Hardwiring on the Remote Manipulator Arm requires an end effector that includes an electrical connector. Connector designs of this type are in the early stages of development and are not flight qualified. At the present time selection of a system using an end effector with an electrical connector could only be predicted on the future qualification of this type of hardware. Because of the apparent complexity of intercenter interfacing in conjunction with the end effector design problems it is felt that this method of hardwiring is the least desirable method for connecting with deployed modules.

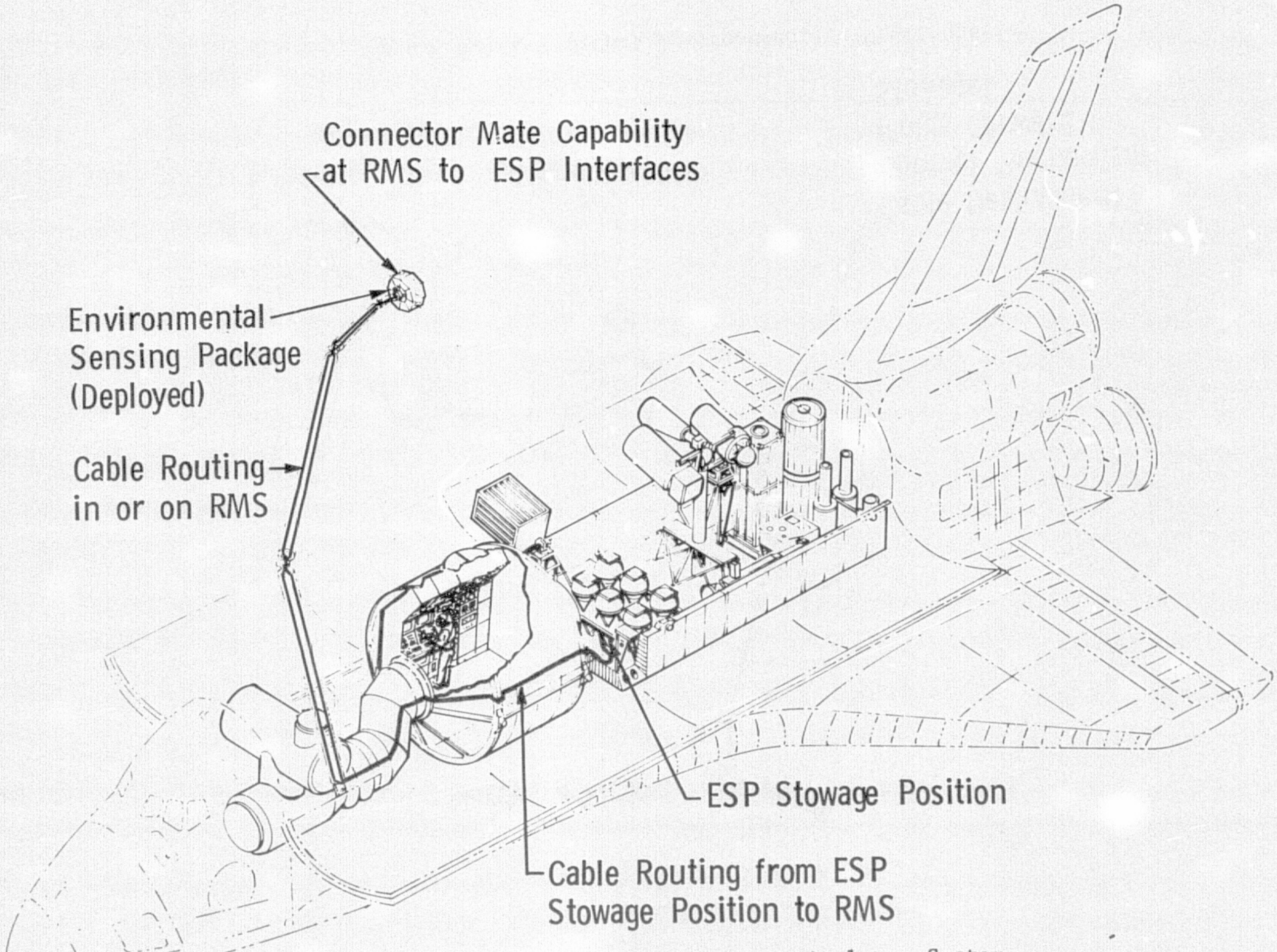


Figure 5.4.1-3 Hardwire, Remote Manipulator System

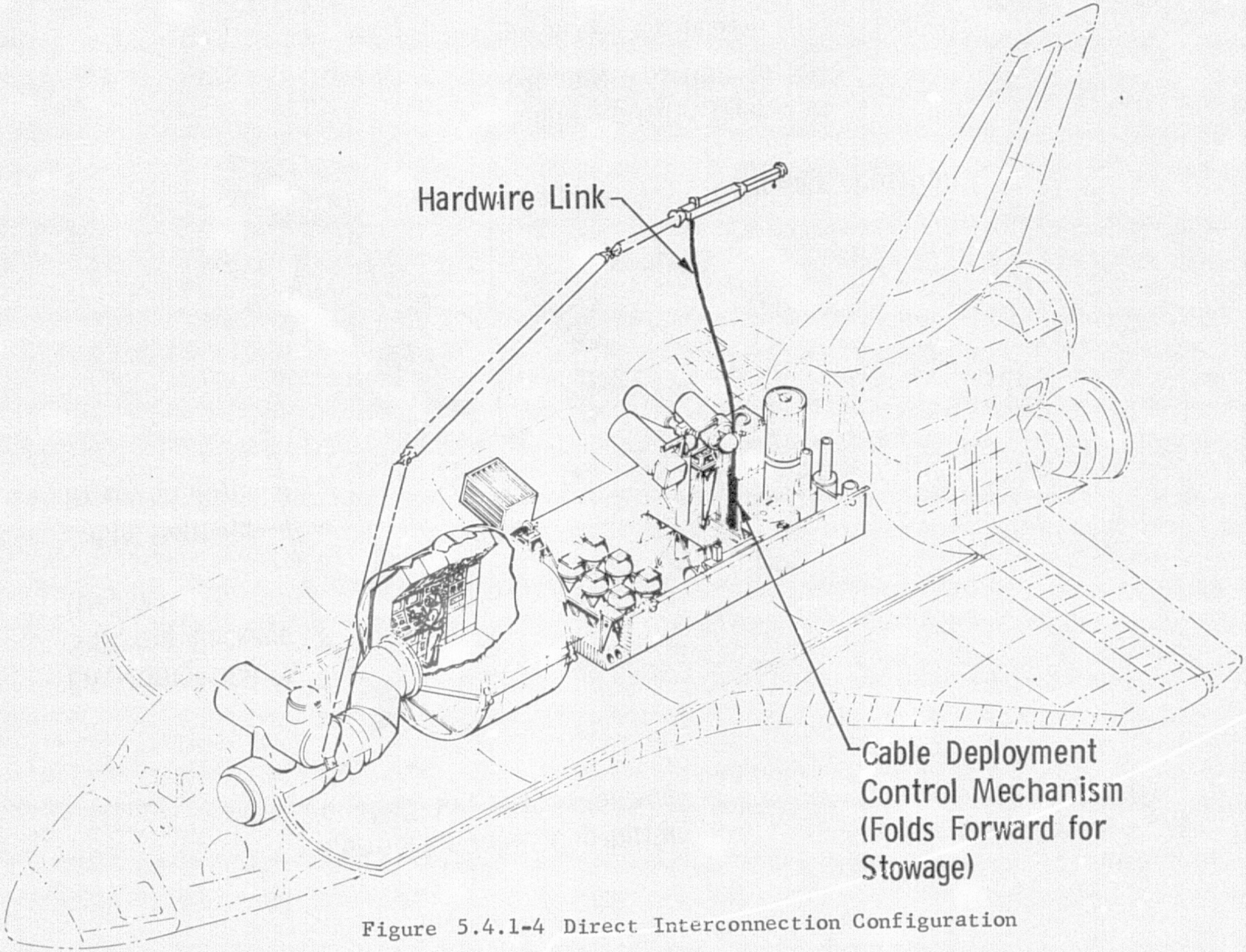


Figure 5.4.1-4 Direct Interconnection Configuration

Figure 5.4.1-4 shows a typical direct pallet to remote module hardwire interconnection. Direct connection to a recoverable module requires a cable of sufficient length to allow the arm to move the modules from one position to another. Cable design must prevent interference with the operation of other instruments either by obstructing their field of view or by mechanical interference.

The cable management system for the direct connection is composed of a cable storage device and a cable guide. Figure 5.4.1-5 shows the configuration of the cable management system considered in this analysis. The cable reel provides a positive force to maintain the cable in a manageable configuration and to retract and stow the cable when the module is returned to the pallet. A survey of existing reel capabilities reveals that a new design would be required for AMPS. Existing designs employ drive mechanisms with switch controls and would require modification for remote control and sensing. Existing configurations are also designed for operation in an earth atmosphere and the effects of space vacuum and temperatures on reel operation would have to be determined by analysis and test.

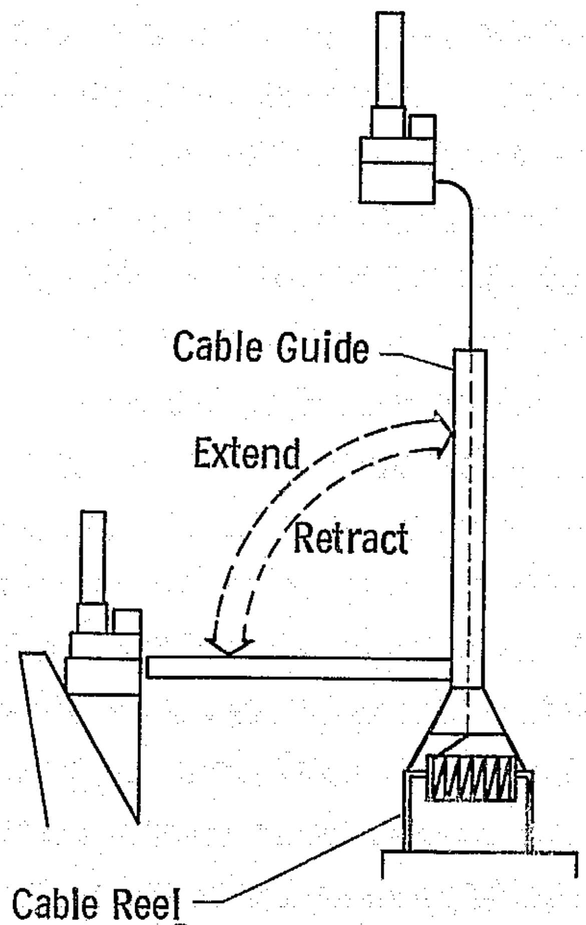


Figure 5.4.1-5 Cable Management System

The cable guide shown in Figure 5.4.1-5 is required to provide a cable pivot point above the payload structure to eliminate interference between the cable and the AMPS instruments. The cable guide design configuration includes the mechanism to drive the guide from the stored to the operational position and return. This guide and the mechanism would be a new design for the AMPS payload. On Flight 1 one guide is required. On Flight 2 two guides are required and the design for different applications may require modification from the Flight 1 configuration.

An additional concern in the hardwire concept is the effect of the environment on the cable. The cable bundle required to support the power, data and command interface includes both twisted shielded pairs and coax cables. The effect of the temperature extremes on the flexibility of the wire is unknown. A recent study on wiring of a similar nature reveals that cable heaters may be required to insure that the cable will extend and retract as required to support the module maneuvering requirements.

RF Transmission - Use of a self-contained power, data and command system for each remote instrument module would require the addition of transmitting and receiving equipment on the module. A self-contained power system would also be required to provide instrument and subsystem operational power while the module is deployed. The design concept for internal data distribution and handling is described in Sections 4.5 and 4.7. This approach, although adding size, weight and complexity to the modules, can take advantage of the free-flying subsatellite RF link and data management design. Figure 5.4.1-2 shows the RF components required to support the Beam Diagnostics Package on Flight 1. The pallet located RF terminal used for communication with the AMPS free-flying subsatellites can be used for all remote modules. Design and development of the RF link approach is cost effective because of the availability of off-the-shelf data handling components.

5.4.1.2 Evaluation of the Candidates

Each of the hardwire configurations involves use of new hardware designed for a particular deployed package. Successful integration of new hardware into a configuration depends on completion of a qualification and test program that demonstrates the capability of the design to meet flight environments. A cursory review of manufacturers of mechanisms of the type required to support the direct connection approach revealed no presently available hardware which could be used. Therefore, a new design, development and qualification program would be required.

The Remote Manipulator Arm routing approach was also considered less desirable because of the new development required for the end-effector connector. Both the high cost of developing such a removable connector and the risk of failure during connect/disconnect operations prove to be disadvantages. The increased complexity of intercenter

and contractor interfaces and added cost for integration with the arm are also factors to be considered.

The self-contained power, data and command system configuration consists of existing, off-the-shelf, flight qualified components. The main disadvantage of this configuration is the addition of weight and volume to the deployed packages. However, the preliminary analysis of the total module weights and volumes reveals that they are acceptable and well within the capability of the Remote Manipulator Arm.

The self-contained system uses the same RF link as the free-flying subsatellites. Commonality of design permits the use of some components on both Flight 1 and Flight 2.

Table 5.4.1-1 is a summary of the advantages and disadvantages of the two candidate configurations.

Table 5.4.1-1 Configuration Design Comparison

Options	Advantages	Disadvantages
Hardwire	Reduced Deployed Weight Direct Connection Less Complex	Effect on RMS Movement Cable Management System Development Cable Heater Required Effect on Field of View of Instruments Increased Interfacing
Self-Contained System	No Effect on RMS Movement Use of Off-the-Shelf Components Component Commonality Between Deployed Packages No Obstruction for Other Instruments	Additional Electronic Packages Increased RMS Deployed Weight

5.4.1.3 Cost

Preliminary cost analysis was completed for the direct hardwire interconnection and the self-contained system. The analysis indicated that the direct hardwire interconnection was three times as expensive as the self-contained system. The main cost difference was in the design, development and qualification of mechanisms required for the hardwire configuration. The self-contained system uses existing design

hardware and therefore only delta qualification is required. Since the hardwire system includes all new design, a great cost risk is associated with that configuration.

The preliminary cost analysis only considered the specific hardware costs. The additional cost of integration with the balance of the payload must also be considered. The integration costs for the RF system would be minimal because the RF interface already requires checkout for free flyers and because no additional physical interfaces would be necessary above attachment of the module to the pallet which is required for both alternatives. The hardwire approach will require mounting of the cable management system, rearrangement of the payload to accommodate the extra equipment, and special testing to assure proper cable clearances during operations. These factors would create a significant increase in integration costs.

5.4.1.4 Configuration Selection

The self-contained power, data and command system has a lower technical risk and a lower cost because of the availability of off-the-shelf hardware. The self-contained system was selected as the baseline configuration for Flight 1 and Flight 2.

5.4.2 Deployed Instrument Power Supply Analysis

The Electrical Power and Distribution System (EPDS) baseline concept for AMPS Flight 1 and Flight 2 includes self-contained power for all deployed instruments. All power for the deployed packages will be provided by silver-zinc batteries of an existing design. The batteries are selected using conservative sizing techniques to insure nominal power and energy increases can be permitted without impacting the baseline design configuration.

Power Supply Selection - The power and energy requirements were computed for each deployed package based on the Flight 1 and Flight 2 timelines. Since the power requirements and the scheduled operating profiles are in a preliminary definition stage, an energy margin of approximately 100% was added to the requirement prior to the selection of the power supply. This results in a very conservative approach that aids the instrument and support subsystem designers in the design and component selection without impacting the baseline design. The Phase C/D design effort will recalculate the energy requirements and provide battery sizes consistent with program design guidelines.

The use of existing silver-zinc batteries for the deployed package power supplies was determined to be the low cost approach. The batteries included in this analysis are qualified for space flight and will not require requalification to the Shuttle environments. The battery size, weight and power dissipation characteristics are consistent with the design guidelines for the deployed packages. The discharge rates are consistent with the rates for cells designed for "low" discharge rates.

The energy requirements for each of the deployed packages on Flight 1 and Flight 2 are given in Table 5.4.2-1. This table shows that five different battery capacities are required to satisfy the energy requirements for the first two AMPS flights. Since the batteries are all existing designs the cost penalty for using a number of different sizes is considered minimal. However, after the final calculations are made for the packages, common batteries will be used wherever possible.

The use of silver-zinc primary batteries to provide power for the AMPS deployed packages offers a highly reliable, low cost design approach. The power margins provided given the design engineer adequate flexibility in defining effective, low cost instruments and support subsystems.

Table 5.4.2-1 Power Supply Summary

FLIGHT/INSTRUMENT PACKAGE	ENERGY REQUIREMENT A-h	BATTERY RATED CAPACITY A-h	BATTERY WEIGHT kg	BATTERY SIZE cm
FLIGHT 1				
Gas Release Module	0.25	1.7	0.74	14 x 6 x 6.5
Environmental Sensing Package	53	105	21.50	36 x 21 x 15
Beam Diagnostic Package	115	160	28.30	40 x 19.5 x 16.5
FLIGHT 2				
Chemical Release Module	0.25	1.7	0.74	14 x 6 x 6.5
RF Receiver Package	26	65	15.64	32 x 17 x 13
Plasma Wake Generator	14	16.8	5.10	22 x 11 x 9.5
Plasma Wake Diagnostic Package	32	65	15.64	32 x 17 x 13

5.5 Attitude and Pointing Control Subsystem

5.5.1 Digital Computer Simulation Model Description

The model used for control system analysis was developed so that, with proper choice of parameters, it can be used to determine the dynamic performance of either the MPM or SIPS for pointing instruments out of the Orbiter payload bay. The model can also be utilized for both inertial and moving targets. The effects on pointing performance due to Orbiter disturbances and to system parameter variations can be evaluated. Figure 5.5.1-1 is a block diagram of the complete model including vehicle three-body model, instrument gimbal system, gimbal drive system, rate gyro dynamics and instrument and torquer control laws.

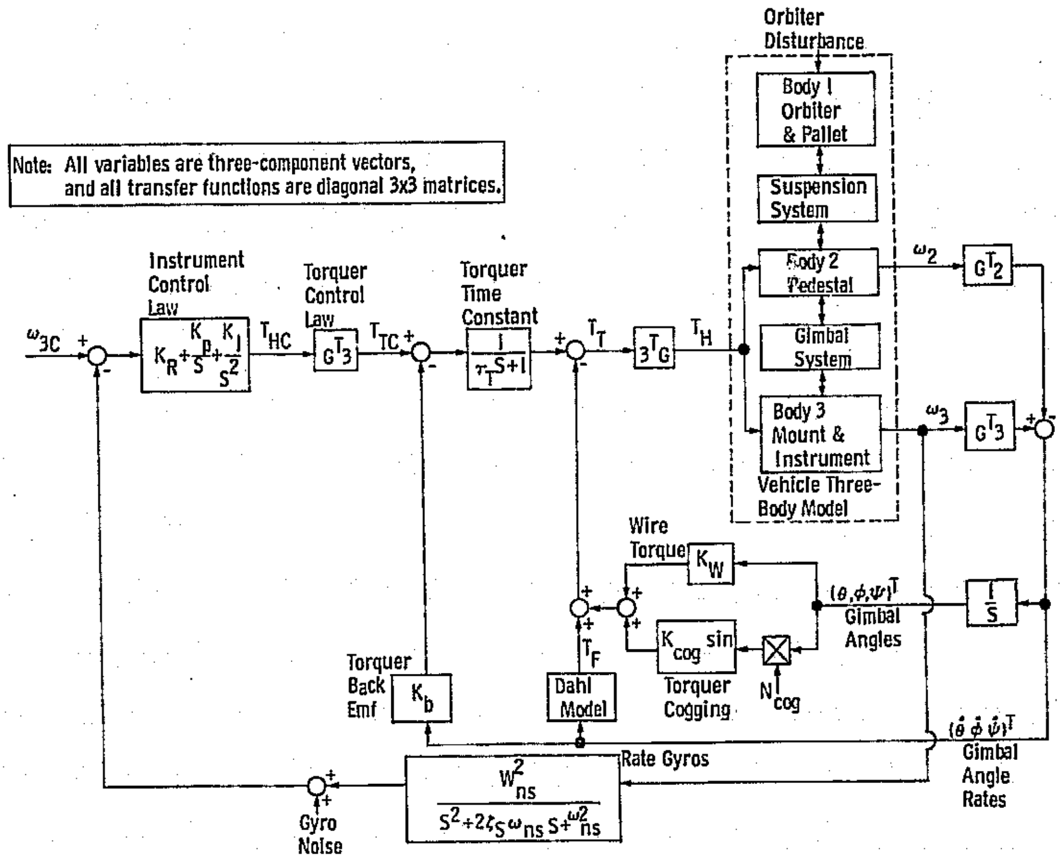


Figure 5.5.1-1 Three-Body Digital Simulation Model

The model consists of the following three bodies: body 1 represents the Orbiter and pallet; body 2 represents the pedestal; and body 3 represents the instrument mount and instrument. (A schematic diagram of the three-body model is shown in Figure 5.5.1-2.) Bodies 1 and 2 are assumed quasi-inertial. Thus, only body 3 nonlinear terms are retained. The pedestal suspension system consists of four identical

three-dimensional linear translational spring dampers contained in a plane perpendicular to the Z axis and symmetrically placed with respect to the X and Y axes passing through the center of elasticity between bodies 1 and 2. The gimbal system is modeled simply as a universal hinge with a controlled hinge torque. A detailed description of the model can be found in Reference 1.

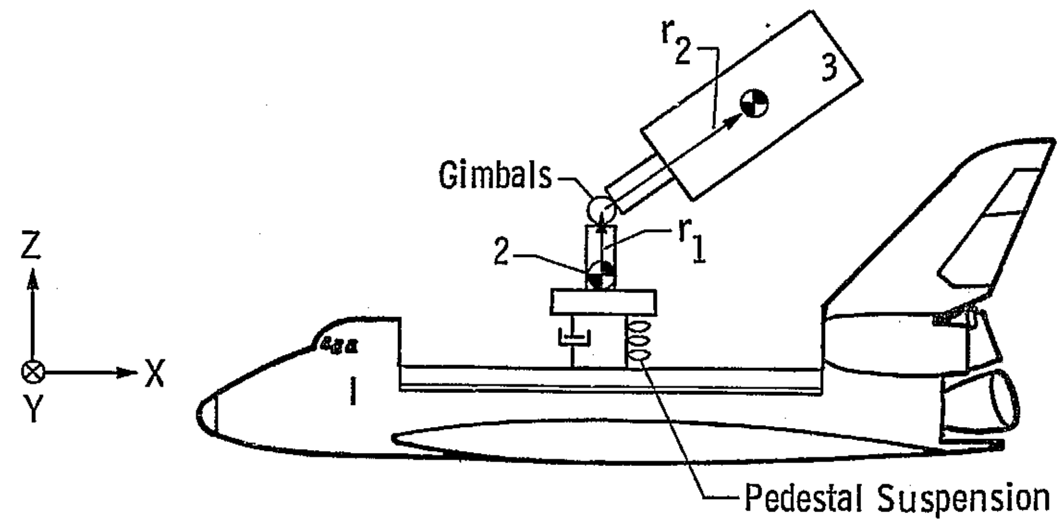


Figure 5.5.1-2 Three-Body Definition

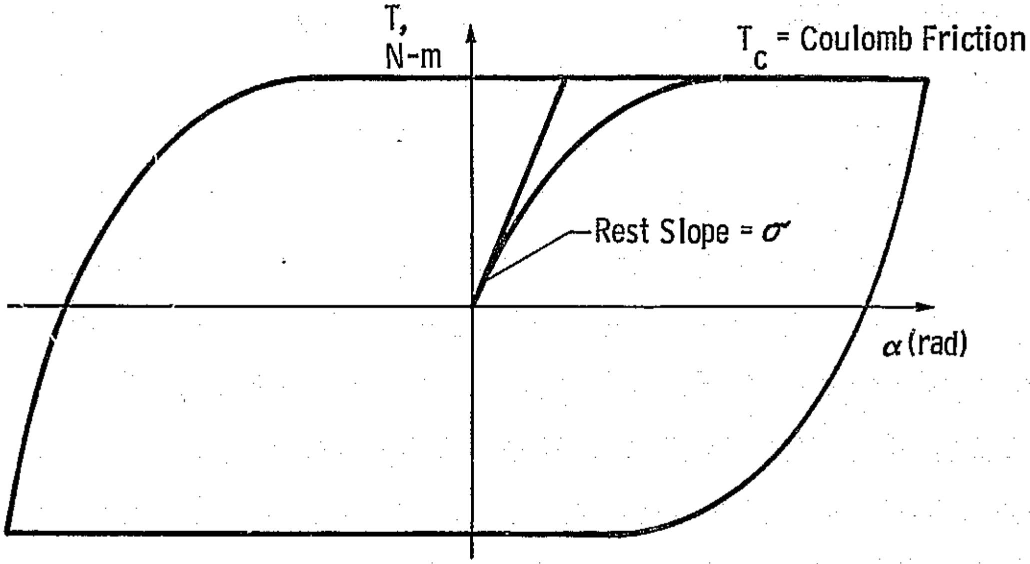
The three-body model was used to study the pointing performance of the MPM and SIPS options as follows: The external torques on bodies 1, 2 and 3 were set to zero. The external forces on bodies 2 and 3 were set to zero, and the external force on body 1 was set to zero or to represent crew motion or thruster firings. To represent the MPM, the suspension parameters between bodies 1 and 2 were set to reflect the MPM pedestal suspension characteristics, and r_2 was set to reflect the instrument center of mass with respect to the gimbal system hinge point. To represent the SIPS, the suspension parameters between bodies 1 and 2 were set to reflect a hard structural interface, and r_2 was set small to reflect the fact that the telescope center of mass is near the gimbal system hinge point.

Reference 1 - "Floated Pallet Definition Study," Final Report, Vol. I, Evaluation of Alternate Telescope Pointing Schemes. George C. Marshall Space Flight Center, Huntsville, Alabama, NASA Contract No. NAS8-30889, May 1975.

Gimbal Drive System - The following gimbal drive system characteristics are included in the model:

- o Gimbal Bearing Friction;
- o Torquer and Resolver Brush Friction;
- o Torquer Magnetic Hysteresis;
- o Torquer Back Emf;
- o Torquer and Instrument Wire Torques;
- o Torquer Time Constant;
- o Torquer Cogging.

The first three of these are simulated either individually or in combination by a Dahl friction model with an exponent equal to one (See Figure 5.5.1-3). Table 5.5.1-1 gives the gimbal drive system parameter values used in this study.



$$\dot{T} = \frac{dT}{dt} = \frac{dT}{d\alpha} \cdot \frac{d\alpha}{dt} = T' \cdot \dot{\alpha}$$

$$T = \int \dot{T}\, dt$$

$$T = \int_0^t \sigma \left| 1 - \frac{T}{T_c} \operatorname{sgn} \dot{\alpha} \right|^{I} \operatorname{sgn} \left(1 - \frac{T}{T_c} \operatorname{sgn} \dot{\alpha} \right) \dot{\alpha}\, dt$$

Reference: AIAA Paper No. 75-1104, August 1975.

Figure 5.5.1-3 DAHL Friction Model

Table 5.5.1-1 Gimbal Drive System Parameter Values

Gimbal Drive System Characteristic		Pointing System MPM	Pointing System SIPS	Units
Gimbal Bearing Friction	σ	1.467	9.737	Nm/rad
	T_c	0.011	0.00141	Nm
Cumulative Friction (Brush Plus Magnetic)	σ	7.05	---	Nm/rad
	T_c	0.02215	---	Nm
Torquer Magnetic Hysteresis	σ	---	54.0	Nm/rad
	T_c	---	0.272	Nm
Torquer Back Emf	K_b	---	0.3296	Nm/rad/s
Wire Torques	K_w	0.0135	0.4	Nm/rad
Torquer Time Constant τ_t	f_n = 1 Hz	0.00796	0.00796	sec
	f_n = 5 Hz	0.00318	0.00318	sec
Torquer Cogging	K_{cog}	0.0035	---	Nm
	N_{cog}	41	---	---

Rate Gyro Dynamics and Noise Characteristics - In order to represent rate gyro dynamic behavior a second-order transfer function of the following form was utilized in each of the X, Y, and Z axes.

$$\frac{\omega_{ns}^2}{s^2 + 2\zeta_s \omega_{ns} s + \omega_{ns}^2}$$

where

ω_{ns} = rate gyro natural frequency

ζ_s = sensor damping ratio

The rate gyro natural frequency was selected for each simulation run in accordance with the control bandwidth utilized. For a 1 Hz control bandwidth ω_{ns} was set at 10 Hz and for a 5 Hz control bandwidth ω_{ns} was set at 50 Hz.

Three different types of rate gyro noise sources were used to evaluate the effects on pointing stability, tracking stability, as well as on control torque.

Rate Gyro Type	Description
A. LDG 540	air bearing or liquid bearing
B. 64 PMRIG	liquid floated
C. Gyroflex	dry type; two-axis rate integrating gyro

These three types of gyros represent major categories of implementation and thus allow for trade-offs in cost and performance. Modelling of their noise characteristics within the simulation models of the MPM and the SIPS resulted in a comparison of the gyros. Since system design requires three rate gyros (X, Y, and Z axes) located on the instrument, three separate independent noise sources were generated for each of the gyro types.

Since the simulation model for the MPM and the SIPS is digital in nature it was necessary to create a file, on magnetic disc, containing discrete samples of the individual gyro noise sources. The general procedure followed is illustrated in Figure 5.5.1-4. White noise with constant power to 300 Hz was transmitted through an appropriate analog filter network (dependent on gyro type) and sampled at a frequency of 1000 Hz before being stored on disc. In order to avoid a folding back of higher frequencies in the power spectral density an eighth-order Butterworth filter was utilized at the output of the analog filter network as illustrated in Figure 5.5.1-5. In this manner 5000 samples of noise were stored for each of the X, Y, and Z axes, for each gyro type. This allows simulation runs of five seconds duration. The system simulation required two noise samples for each integration interval and thus the noise was sampled from the disc at 200 and 1000 Hz, respectively, for integration intervals of 0.01 and 0.002 seconds.

A description of the analog program used to generate these noise sources is also shown in Figure 5.5.1-5. The analog filter networks utilized to shape the power spectral densities of the LDG 540, the Gyroflex, and the 64 PMRIG are described in detail.

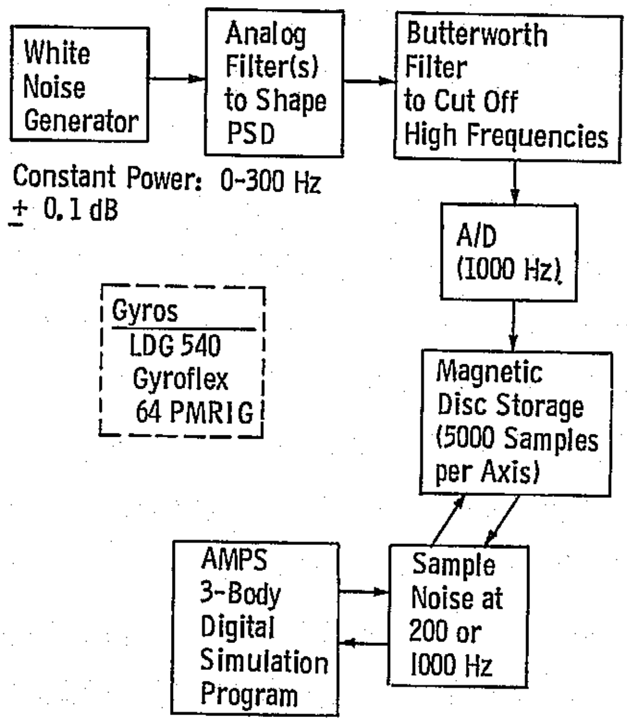


Figure 5.5.1-4 Method of Simulating Gyro Noise

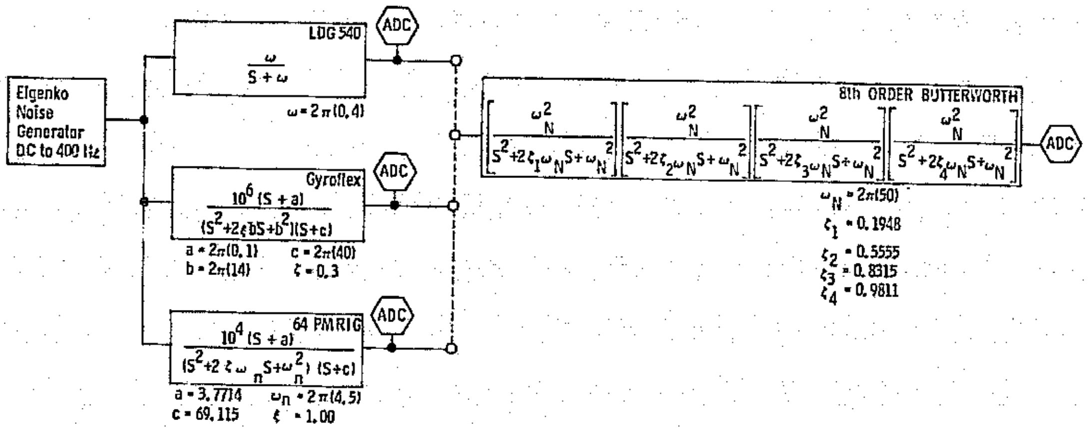


Figure 5.5.1-5 AMPS Noise Filters

Instrument and Torquer Control Laws - Since instrument attitude errors are small during pointing, the instrument attitude reference system is simply the instrument control law (rate plus position plus integral of position) and the inverse torquer control law.

For low frequencies (i.e., $S \approx 0$), the MPM or SIPS pointing system characteristic equation is approximately $[J_3 S^2 + K_R S + K_P]$ assuming K_I is small. Thus, it follows that

$$K_R = 4\pi J_3 \zeta f_n$$
$$K_P = 4\pi^2 J_3 f_n^2$$
$$K_I = 4/5\, \pi^3 J_3 f_n^3$$

describe realistic values for instrument control law gains, where f_n is the desired system natural frequency and ζ is the desired system damping ratio. The calculated instrument control law gain values used in the simulation are listed in Table 5.5.1-2.

Table 5.5.1-2 Instrument Control Law Gains

MKS Units $\zeta = 0.707$		Light Instrument			Heavy Instrument		
		X	Y	Z	X	Y	Z
$f_n = 1$ Hz	K_R	1.53×10^3	1.53×10^3	1.75×10^2	4.82×10^3	4.82×10^3	6.73×10^2
	K_P	6.79×10^3	6.79×10^3	7.79×10^2	2.14×10^4	2.14×10^4	2.99×10^3
	K_I	4.27×10^3	4.27×10^3	4.89×10^2	1.34×10^4	1.34×10^4	1.88×10^3
$f_n = 5$ Hz	K_R	7.65×10^3	7.65×10^3	8.77×10^2	2.41×10^4	2.41×10^4	3.36×10^3
	K_P	1.70×10^5	1.70×10^5	1.95×10^4	5.35×10^5	5.35×10^5	7.47×10^4
	K_I	5.33×10^5	5.33×10^5	6.11×10^4	1.68×10^6	1.68×10^6	2.34×10^5

Model Excitations - Two Orbiter disturbance input profiles and one instrument rate command profile were utilized to determine MPM and SIPS performance. The two Orbiter disturbance profiles used were:

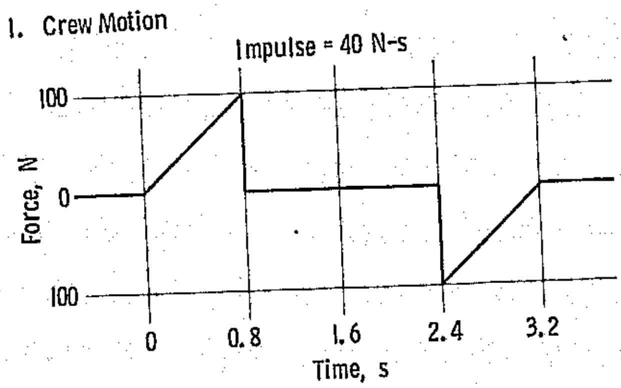
(1) Crew motion in the forward portion of the Orbiter, Moment Arm = $-15.1\,\bar{i} - 0.04\,\bar{k}$ (m);

(2) Thruster firing in Z direction at Orbiter aft,
Moment Arm = 17.4 $\bar{i}$ (m).

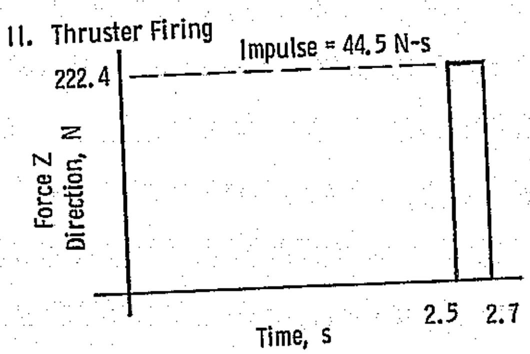
These disturbances are given in Figure 5.5.1-6. The instrument rate command profile used represents the rate required to track an earth fixed point from an altitude of approximately 350 nm with the Orbiter in the Z local vertical Y perpendicular to the orbital plane attitude. This is given by the equation:

$$\omega_{3C} = -1.748 \times 10^{-2} \operatorname{sech}^2 \left(\frac{372-t}{65} \right)$$

Time t = 0 is the time at which the target first becomes visible over the horizon. Time t = 744 seconds is the last time the target is in view. Due to the computer running time involved it is not feasible to run the complete profile. Thus the computer was initialized for a $\pm$2.5 second profile run about the point at which maximum rate occurs, t = 372 seconds.



Crew Wall Pushoff Disturbance Profile



Note: 1. Represents effect of pair of pitch thrusters on orbiter aft.
2. Lever arm: 17.392 m (X direction).
3. Resultant torque: T_y = -3869.0 N-m.

Figure 5.5.1-6 Crew Motion and Pitch Thruster Firing Force Profiles

5.5.2 MPM Preliminary Pointing Performance Analysis

A preliminary pointing performance analysis was performed on the MPM in order to select values for some of the system parameters prior to a detailed performance analysis. The factors whose effects were investigated initially consisted of instrument mass, instrument pointing inclination (look angle), isolator characteristics, and control loop bandwidth. By investigating the effects of these factors in the preliminary study, the detailed performance study could be concentrated on evaluating the effects of various rate gyro noise sources and friction levels. The simulation model utilized for the preliminary performance analysis was the three-body model as defined in Section 5.5.1. Sensor noise and bearing frictions were not included in the simulation. The MPM was used for the preliminary analysis because of available data and the applicability of some results to the SIPS platform.

Discussion - This analysis was performed to obtain a preliminary indication of the effects of instrument look angle, control loop bandwidth, instrument mass, and isolator spring and damping characteristics. The parameters utilized for the study are summarized in Table 5.5.2-1. A set of 34 computer simulation runs was performed at the particular conditions indicated in the table using bandwidths of 1 Hz and 5 Hz and rotational damping ratios of 0.1 and 0.01. Each run was of five seconds real time duration. The instrument was attitude initialized in accordance with the particular look angle indicated in the table. The error generating perturbations expected during a pointing sequence are due to crew motion. During each run an X-axis (positive X-direction: forward to aft in the Shuttle) crew motion disturbance was executed in accordance with the NASA standard force profile shown in Section 5.5.1. The beginning of the force profile was aligned with the beginning of each simulation run. One additional run was used to investigate the effects of a force profile due to the pitch thruster firing. This profile is also illustrated in Section 5.5.1. The thruster force was executed from 2.5 to 2.7 seconds after the start of the run.

The output variables studied in each run consisted of pointing errors and motor torque required. The time history of the pointing errors in each of three axes were scanned to determine the maximum error observed. The maximum stability error observed in any axis and the maximum motor torque were then determined and utilized in the presentation of the results. The maximum torque results can be compared to the maximum available value for the assumed motor (0.64 Nm).

Table 5.5.2-1 Combinations of Study Parameters Utilized

Instrument Look Angle, θ (Degrees)		0						30						60						90					
Isolator Characteristics		S1		S2		HM		S1		S2		HM		S1		S2		HM		S1		S2		HM	
Instrument Mass		L	H	L	H	L	H	L	H	L	H	L	H	L	H	L	H	L	H	L	H	L	H	L	H
Control Loop Bandwidth (Hz)	1	X	X	X	X	X	X					X						X		X	X	X	X	X	X
	2					X	X																		
	5	X	X	X	X	X	X													X	X	X	X	X	X
	10					X	X																		

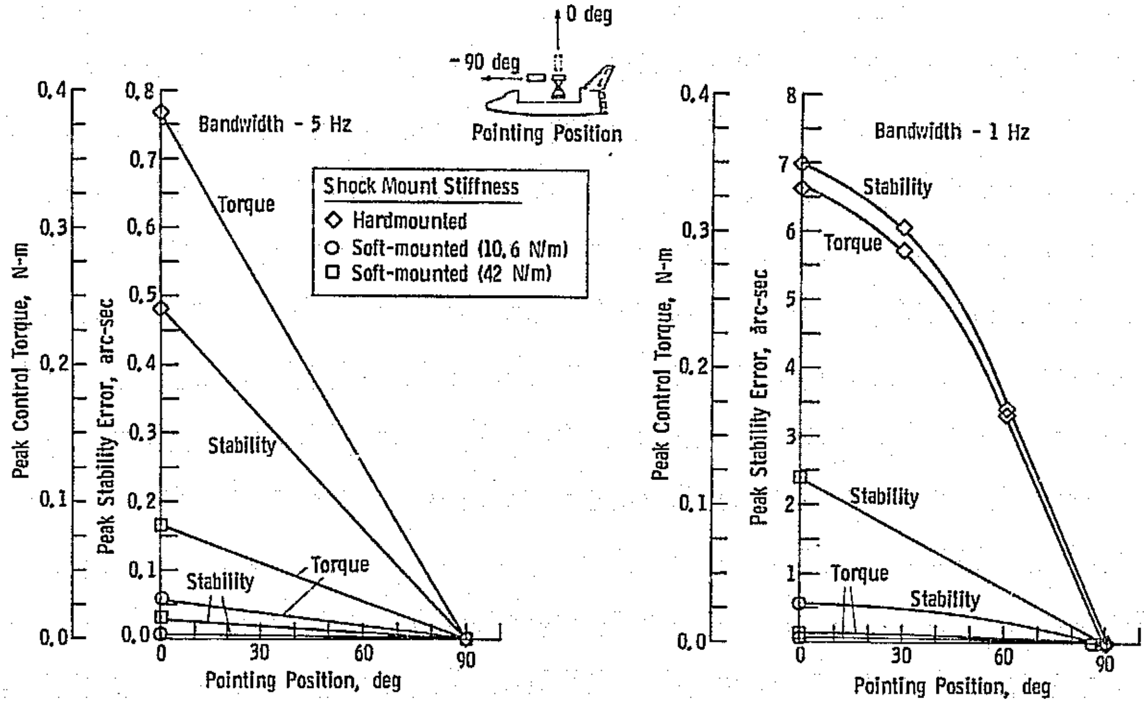
Additional Simulation Runs: a. θ = 90 deg, HM, L, crew motion in Z axis

b. θ = 0 deg, HM, L, pitch thruster disturbance

Notes: L - Light
H - Heavy
S1 - Soft
S2 - Intermediate
HM - Hardmounted

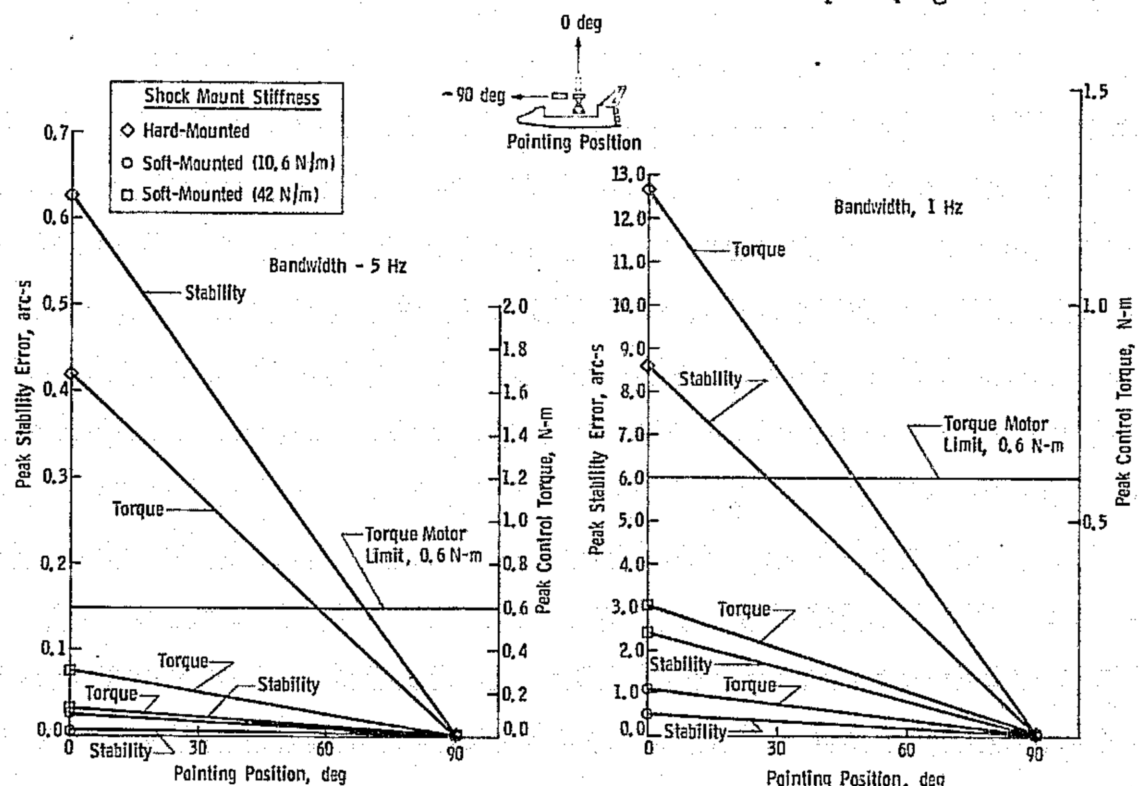
<u>Results</u> - For each simulation run, the maximum stability (pointing) error and the maximum control torque observed in any axis was determined by inspection of the time history printouts of the program. These variables were then used to establish the effects of varying system parameters.

The effects of pointing position (look angle), isolator stiffness, and control bandwidth on peak stability error and peak control torque are illustrated in Figures 5.5.2-1 and 2. Both light and heavy instruments were considered. The left-hand graph of the figures contain the results obtained at 5 Hz control bandwidth, while the right-hand graph of each figure contains the results at 1 Hz control bandwidth. Each graph contains a plot of both the peak stability error and the peak control torque. Effects due to different isolator stiffness are indicated by the various families of curves in each graph. The results obtained for the two additional simulation runs described in Table 5.5.2-1 are given in Table 5.5.2-2. Table 5.5.2-3 demonstrates the effect of varying the rotational damping ratio and control bandwidth.



Instrument Mass: Basic Instrument - 52 kg; with Support Equipment - 167.5 kg

Figure 5.5.2-1 MPM Stability Error and Control Torque VS Pointing Position for Crew Disturbance Input (Light Instrument)



Instrument Mass: Basic Instrument - 500 kg; with Support Equipment - 615.5 kg

Figure 5.5.2-2 MPM Stability Error and Control Torque VS Pointing Position for Crew Disturbance Input (Heavy Instrument)

Table 5.5.2-2 Additional Simulation Runs (a) and (b) Results

Run	Type of Mount	Control Bandwidth (Hz)	Pointing Inclination (deg)	Disturbance Input	Maximum Motor Torque (N-M) (1)	Peak Stability Error (arc-s)
a	Hard-Mounted	1	-90	Z-Axis Crew Motion	0.226	4.50
b	Hard-Mounted	1	0	Pitch Thruster Firing	0.00021	4.04

(1) Typical MPM Motor Torque Capability - 0.64 N-M

Table 5.5.2-3 MPM Control Bandwidth and Isolator Damping Ratio Evaluation (Pointing)

Run	Rotational Damping Ratio	Control Bandwidth, Hz	Rotational Natural Frequency, Hz	Extension X-Axis, mm	Extension Z-Axis, mm	Peak Stability Error (Y), arc-s	Control Torque, N-M
1	0.1	1	0.075	0.448	0.964	0.49	0.0095
2	0.1	5	0.075	0.446	1.04	0.0067	0.0093
3	0.01	1	0.075	-	-	0.53	0.10
4	0.01	5	0.075	-	-	0.0076	0.10

Note: 1. Heavy instrument mass - 615 kg.
2. Crew motion disturbance.

Conclusions - The following conclusions can be drawn by inspection of the study results.

(1) Effect of Control Bandwidth - Peak stability pointing error as well as peak control torque are improved by an order of magnitude by increasing control bandwidth from 1 to 5 Hz for all isolator mounts. Stability is in the sub arc-second range for both bandwidths for the soft-mount (spring constant of 10.6 N/m). The improvement factor is about the same for both instrument mass values studied.

(2) Effect of Instrument Pointing Position (Look Angle) - Peak stability error and peak control torque are greatest when the instrument is pointed straight up, and are smallest when the instrument is aligned with the longitudinal axis of the Shuttle. This result is evidenced by the fact that for the MPM, an X direction force has the greatest lever arm about the instrument center of mass (CM) when the

instrument is pointing upward; and proportionately less when the CM of the instrument is aligned with the hinge point in the X direction.

(3) Effect of Instrument Mass - In some instances the stability pointing error increased by about 20 percent for the heavy instrument as opposed to the light instrument at both bandwidths. The required torque for the heavy instrument was several times that for the light instrument. The torque motor limit for the MPM was exceeded only for the heavy instrument hard-mounted at some range of the pointing position near vertical pointing.

(4) Effect of Isolator Stiffness - Stability pointing error and control torque decrease significantly as the stiffness of the isolators is reduced. Sub arc-second stability is achieved for both bandwidths and both instrument mass values for the softest isolator examined.

(5) Isolator Rotational Damping Ratio - The difference in peak stability between the two damping ratios utilized (0.1 and 0.01) was found to be insignificant. Peak control torque observed was one order of magnitude greater for the damping of 0.01.

Evaluation of the effects of the five factors on pointing stability and control torque led to a reduction of the number of parameters to be considered for detailed analysis. The parameters selected are listed below along with the rationale for selection.

(1) Control Loop Bandwidth - 1 and 5 Hz; satisfactory performance was achieved for a set of soft isolators.

(2) Isolator Stiffness - $K_x = K_y = 8$ N/m, $K_z = 28$ N/m; increased stability can be achieved through the use of softer isolators.

(3) Damping Ratio Value - 0.01; damping ratio value has little effect on pointing stability.

(4) Instrument Mass - Heavy Instrument; pointing stability degradation and increased motor torque indicated a worst case for the heavy instrument.

(5) Instrument Look Angle - 0^o (straight up); pointing stability degradation and increased motor torque were evident at this angle.

5.5.3 Detailed Pointing Performance Analysis

The objective of this study was to establish the effects of rate gyro noise and various friction levels on pointing performance of the SIPS and MPM. A number of parameters selected for the analysis were as specified in Section 5.5.2. The three-bodv model, described in Section 5.5.1, was utilized for all simulation runs and was updated to include models of various types of friction, hysteresis effects, and rate gyro noise as they applied to the two pointing platforms. Performance was evaluated in terms of pointing stability and required gimbal motor torque.

Discussion - Each pointing platform was oriented so that the instrument axis was perpendicular to the pallet floor. The effects were observed for perturbations about this position caused by wall push-off motion in the forward portion of the Orbiter. The time frame of the crew motion profile, described in Section 5.5.1, was aligned with the beginning of each simulation run. The model was modified to include rate gyro noise as well as pertinent bearing frictions and motor magnetic effects.

The rate gyros considered in this study consisted of the LDG540, the Gyroflex, and the 64PM RIG. The techniques utilized to simulate their noise characteristics (power spectral densities) are given in Section 5.5.1. The noise signal, properly scaled, was added to the output signal of the rate gyro in each of three axes (X,Y,Z). The same model of rate gyro was used in each axis for a given run. The noise signals for the three axes were statistically independent of each other. Position sensors were considered as ideal for this study leading to calculation of the position signal by integrating the rate signal with no noise.

The hardware frictional and other resistance-type characteristics simulated for the SIPS and the MPM are summarized in Table 5.5.1-1. Bearing frictions and magnetic hysteresis were simulated in the form of a Dahl-type model. Slope at the origin, an exponent, and a Coulomb value are given for each. Cable torques were considered for the SIPS and was simulated by means of a spring constant. The effect of increasing friction beyond the nominal value was investigated for both the SIPS and the MPM. Since a Dahl model was used in each case, both the slope and the Coulomb level were increased by the same factor for the purpose of this analysis. This procedure was followed for the cumulative Dahl and the main bearing Dahl friction for the MPM as well as for the motor magnetic hysteresis and the Dahl bearing friction for the SIPS.

Results - The conditions (parameter values) under which each run was made are specified below. Only the non-nominal parameters and/or the parameters specifically under study are specified.

- o look angle: 0 deg (straight up)
- o isolators (MPM only): soft isolators ($K_x = K_y$ = 8N/m, K_z = 28N/m; rotational damping ratio of 0.01)
- o crew motion disturbance starting at time t = 0.0
- o heavy instrument (500 kg plus support equipment)
- o The control bandwidth, the rate gyro noise source for all three axes, and the type of friction utilized varied as noted for the simulation runs.

The largest friction characteristic considered for the SIPS was the motor magnetic hysteresis. For the MPM it was the cumulative Dahl friction (combined effect of encoder bearing friction, brush friction, and motor magnetic hysteresis). The effect on pointing stability due to these frictions, the three types of rate gyro noise, and control bandwidth are illustrated in Tables 5.5.3-1 and 5.5.3-2 for the SIPS and MPM, respectively. The combined effect of these error sources is also given.

Table 5.5.3-1 SIPS Pointing* Results

Run No.	Control Bandwidth, Hz	Rate Gyro Noise Source	Type of Friction	Maximum Stability Pointing Error, arc-s	Maximum Motor Torque N-m
1	1	None	None	0.016	0.002
2	1	LDG 540	None	0.015	0.002
3	1	64 PMRIG	None	0.016	0.002
4	1	Gyroflex	None	0.016	0.015
5	1	LDG 540	Dahl**	0.402	0.006
6	1	64 PMRIG	Dahl	0.404	0.006
7	1	Gyroflex	Dahl	0.40	0.014
8	5	Gyroflex	Dahl, Cable Torques	0.010	0.064
9	5	LDG 540	Dahl, Cable Torques	0.0088	0.005
10	5	64 PMRIG	Dahl, Cable Torques	0.010	0.005

* Crew motion disturbance; nominal system.
** Represents motor magnetic hysteresis.

Table 5.5.3-2 MPM Pointing* Results

Run No.	Control Bandwidth, Hz	Rate Gyro Noise Source	Type of Friction	Maximum Stability Pointing Error, arc-s	Maximum Motor Torque, N-m
1	1	None	Cumulative Dahl**	0.497	0.10
2	1	LDG 540	None	0.47	0.09
3	1	Gyroflex	None	0.468	0.10
4	1	64 PMRIG	None	0.47	0.09
5	1	LDG 540	Cumulative Dahl	0.499	0.10
6	1	Gyroflex	Cumulative Dahl	0.501	0.11
7	1	64 PMRIG	Cumulative Dahl	0.499	0.10
8	5	LDG 540	None	0.008	0.10
9	5	Gyroflex	None	0.016	0.20
10	5	64 PMRIG	None	0.007	0.10

* Crew motion disturbance; nominal system.

** Represents combined effect of encoder bearing friction, brush friction, and motor magnetic hysteresis.

The effect of MPM control bandwidth is illustrated in Figure 5.5.3-1 for all three types of rate gyros. Note that a nominal system was utilized; i.e., no friction inputs.

The individual and cumulative effects on pointing stability of a number of rate gyro noise and friction conditions are indicated in Table 5.5.3-3. The results are subdivided into three categories as a function of the type of disturbance force profile executed. For both pointing platforms the worst-case friction input was used (SIPS, motor magnetic hysteresis; MPM, cumulative Dahl friction).

The effect of increasing and decreasing the frictional values from nominal is illustrated in Figures 5.5.3-2 and 5.5.3-3 for the SIPS and the MPM, respectively. Note that the horizontal axis of each graph gives the normalized value of friction, i.e., the multiples of nominal friction used in each case. In deviating from nominal both the slope and the Coulomb level were multiplied by the given normalizing factor.

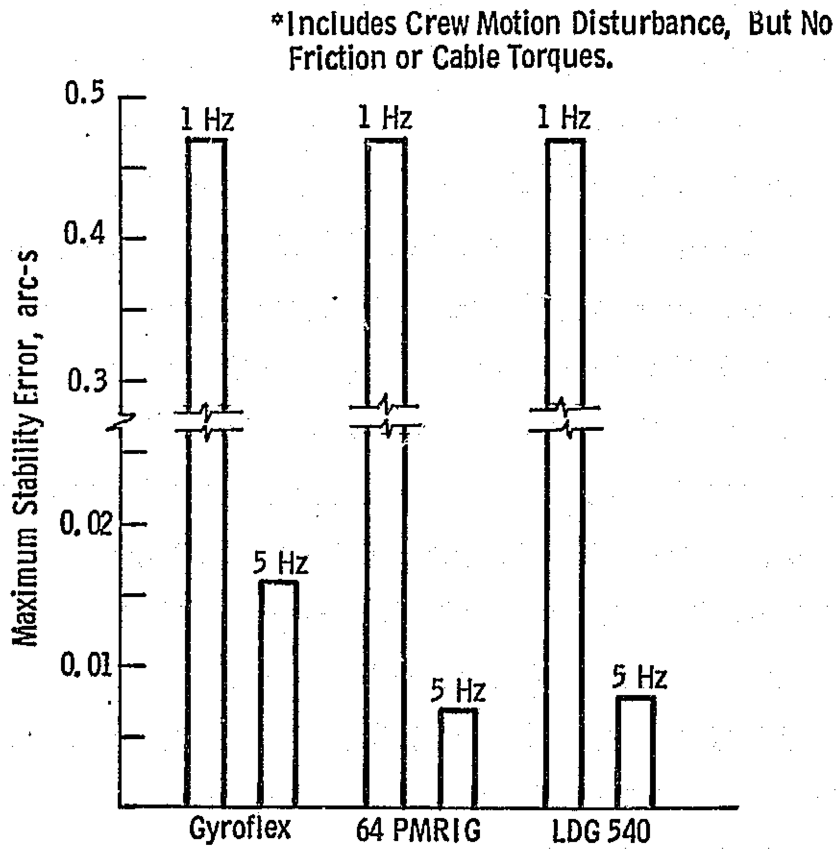


Figure 5.5.3-1 MPM Pointing Stability* Vs Control Bandwidth

Table 5.5.3-3 Individual/Cumulative Disturbance Effects on Pointing Stability

Case	Condition	SIPS, arc-s	MPM arc-s
I. Crew Motion	Crew Motion Alone	0.016	0.470
	Cable Torque	0.016	0.468
	Friction*	0.404	0.497
	LDG 540	0.015	0.47
	64 PMRIG	0.016	0.47
	Gyroflex	0.0153	0.47
	LDG 540, Friction, Cable Torque	0.406	0.498
	64 PMRIG, Friction, Cable Torque	0.406	0.498
	Gyroflex, Friction, Cable Torque	0.404	0.496
II. Crew Motion with Thruster Firings	LDG 540, Friction, Cable Torque	0.404	--
	64 PMRIG, Friction, Cable Torque	0.406	--
	Gyroflex, Friction, Cable Torque	0.404	--
III. Thruster Firings	Thruster Firing Alone (No Crew Motion)	0.022	--

* Friction utilized:
SIPS - motor magnetic hysteresis,
MPM - cumulative Dahl.

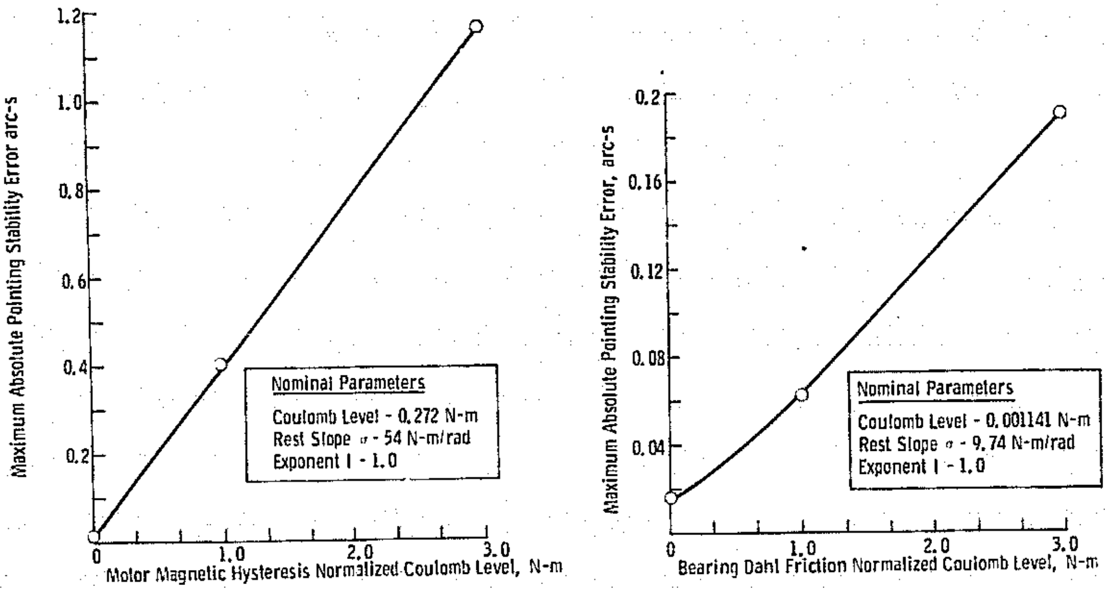


Figure 5.5.3-2 Evaluation of Friction Effects on SIPS Stability Under Crew Motion

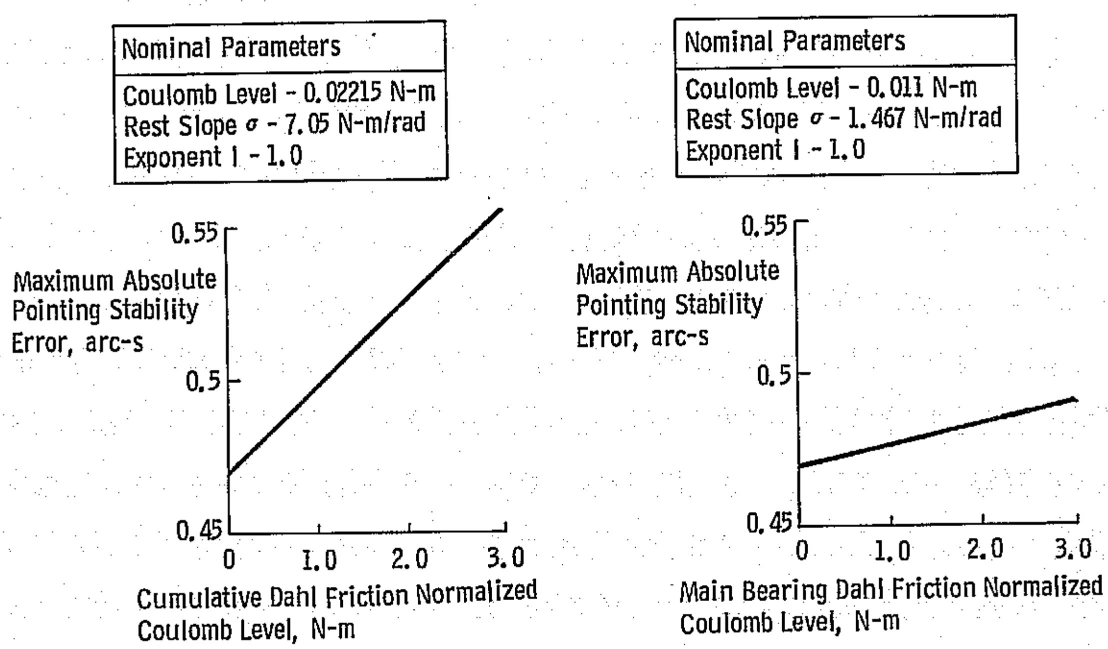


Figure 5.5.3-3 Evaluation of Friction Effects on MPM Stability Under Crew Motion

Conclusions - The pointing study revealed that both systems, SIPS and MPM, are capable of sub arc-sec pointing stability even when crew motion activity takes place and when rate gyro noise and worst-case hardware friction are included. The pointing stability of the SIPS and the MPM are about 0.4 and 0.5 arc sec respectively under these conditions for a 1 Hz control loop bandwidth. Maximum motor torques are similarly well below available output. By increasing the control bandwidth to 5 Hz the pointing stability of both systems is improved by more than an order of magnitude.

It was also found for both systems that the effect of adding noise to the rate gyro outputs is essentially negligible for the noise levels investigated. In addition, the results indicate that differences between rate gyro models has a negligible effect on pointing stability. Differences between rate gyro models did, however, have a more pronounced effect on maximum motor torque exerted as shown by the results.

The effect of adding the various types of friction was found to have a significant effect on both systems in comparison to cases in which friction was not considered. As expected, the effect of adding friction also significantly increased the required amount of motor torque. By inspection of Figures 5.5.3-2 and 5.5.3-3 it is evident for either system that an increase in any friction torque results in a higher pointing stability error. The relationship between the magnitude of friction and the pointing stability error appears approximately linear.

Thruster firings appeared to have about the same effect on stability as crew motion, when acting as separate disturbance inputs.

5.5.4 Tracking Performance Analysis

Some proposed AMPS experiments require a pointing platform capable of earth tracking, continuously pointing an instrument at a point which is fixed relative to the earth's surface over an extended period of time (typically 15 minutes). The target may be a point on the earth's surface (as in remote sensing applications) or a point fixed relative to the earth at a fixed distance above the earth's surface (as in atmospheric studies or stellar occultation measurements).

The tracking rate and position command profiles were based upon a knowledge of target coordinates in an earth-fixed system and data on Shuttle location and orientation relative to the earth. Pointing errors were measured with respect to these profiles.

Discussion - A series of tracking simulations was performed to determine the relative performance of the SIPS and MPM in this mode. The tracking command profiles which represent the rate and attitude required to track an earth-fixed point were supplied by NASA. They have the following form:

$$\theta(t) = 1.136 \tanh\left(\frac{372-t}{65}\right)$$

$$\omega(t) = (-1.7476923 \times 10^{-2}) \operatorname{sech}^2\left(\frac{372-t}{65}\right)$$

In all tracking runs, the instrument was slewed about the Orbiter -Y axis; $\theta(t)$ is the angle the instrument is rotated about the slewing axis at time t; the angular velocity is denoted by $\omega(t)$. Since t is in seconds, the duration of the tracking profile is 744 seconds and θ takes on values between +65 and -65 degrees. The tracking profile for θ is given in units of radians.

Due to the computer running times involved it was not feasible to run a complete tracking profile. In order to achieve realistic computer running times, at first some runs were of twenty seconds duration, and the majority of runs were of 5 seconds duration. The system was initialized properly for ± 10 or ± 2.5 seconds, respectively, about the point at which maximum rate occurs directly over the target region. This method of initialization thus contained the time of maximum rates, attitude errors and isolator extensions.

Results - As in the pointing simulation study, the motor torque required is, however, a function of friction and rate gyro noise. In the case of the MPM the LDG 540 and the 65 PMRIG rate gyros have negligible effect on the maximum motor torque required. For these two gyros with friction included in the model, the maximum resultant motor torque is 0.0437 newton-meters; for the Gyroflex including friction, the maximum motor torque is 0.0584 newton-meters. It should be noted that while the "middle" 5 seconds of the tracking profile include the maximum experiment rates, the maximum accelerations occur at around t = 342 seconds, or 30 seconds before the "middle" of the profile.

The SIPS results are very similar. The tracking stability error is independent of the rate gyro, and amounts to 1.48 arc sec. The stability error with no gyro noise and no friction included is 1.24 arc sec. In all cases where friction is included, the stability error is 2.49 arc sec and is independent of the gyro noise source. In these cases, the maximum motor torque is nearly constant at 0.29 newton-meters. For the Gyroflex and no friction, the maximum motor torque required is higher than that of the LDG 540 and the 64 PMRIG. The MPM and SIPS tracking results are summarized in Tables 5.5.4-1 and 5.5.4-2, respectively. Since the SIPS is hard mounted the isolator extensions do not apply.

Table 5.5.4-1 MPM Tracking* Results

Run No.	Control Bandwidth, Hz	Rate Gyro Noise Source	Type of Friction	Maximum Stability Pointing Error, arc-s	Maximum Motor Torque, N-m	Maximum Isolator Extension, mm
1	1	None	None	1.35	0.0213	0.213
2	1	None	Cumulative Dahl**	1.33	0.0434	0.213
3	1	LDG 540	None	1.34	0.0213	0.213
4	1	64 PMRIG	None	1.36	0.0214	0.213
5	1	Gyroflex	None	1.36	0.0333	0.214
6	1	LDG 540	Cumulative Dahl	1.33	0.0436	0.214
7	1	64 PMRIG	Cumulative Dahl	1.33	0.0437	0.214
8	1	Gyroflex	Cumulative Dahl	1.34	0.0584	0.213

*Earth target tracking for 5 seconds real time at time of highest instrument tracking rate; no crew disturbances, nominal system.

**Represents combined effect of encoder bearing friction, torquer brush friction, magnetic hysteresis.

Table 5.5.4-2 SIPS Tracking* Results

Run No.	Control Bandwidth, Hz	Rate Gyro Noise Source	Type of Friction	Maximum Stability Pointing Error arc-s	Maximum Motor Torque N-m
1	1	None	None	1.24	0.015
2	1	None	Dahl**	2.49	0.282
3	1	LDG 540	None	1.48	0.015
4	1	64 PMRIG	None	1.47	0.015
5	1	Gyroflex	None	1.48	0.021
6	1	LDG 540	Dahl	2.49	0.283
7	1	64 PMRIG	Dahl	2.49	0.283
8	1	Gyroflex	Dahl	2.49	0.290

*Earth target tracking for 5 seconds real time at time of highest instrument tracking rate; no crew disturbances, nominal system.

**Represents motor magnetic hysteresis.

Conclusions - For tracking under realistic conditions (rate gyro noise and hardware friction) the maximum pointing stability error of the MPM is 1.36 arc sec and the SIPS is 2.5 arc sec. Isolator extensions and motor torques for the MPM are within the acceptable maxima of 0.0025 meters and 0.6 newton-meters, respectively. Either platform can meet the AMPS tracking requirements but it should be noted that neither the SIPS nor the MPM achieved sub arc sec tracking stability error.

5.5.5 Preliminary Static Budget

A preliminary static error budget study was conducted in order to determine which component errors would be likely to contribute to static pointing errors and which component trade-off options are beneficial to the system designer. The results of the study are given in such a form that when specific system components become defined or a change in components is anticipated, a quick reference to the results given here can yield an estimate of the effect on static pointing accuracy.

The generalized system block diagram illustrated in Figure 5.5.5-1 includes only the major system components relevant to the static error budget model. From this diagram the error equation of the system attitude was derived and subsequently the variance of the pointing error. The major error sources were attributed to quantization level errors in the star tracker and rate gyro signals, rate gyro drift, and star tracker accuracy. For derivation of the error model the system was assumed to be in steady state; i.e., no control torque is being requested by the control laws.

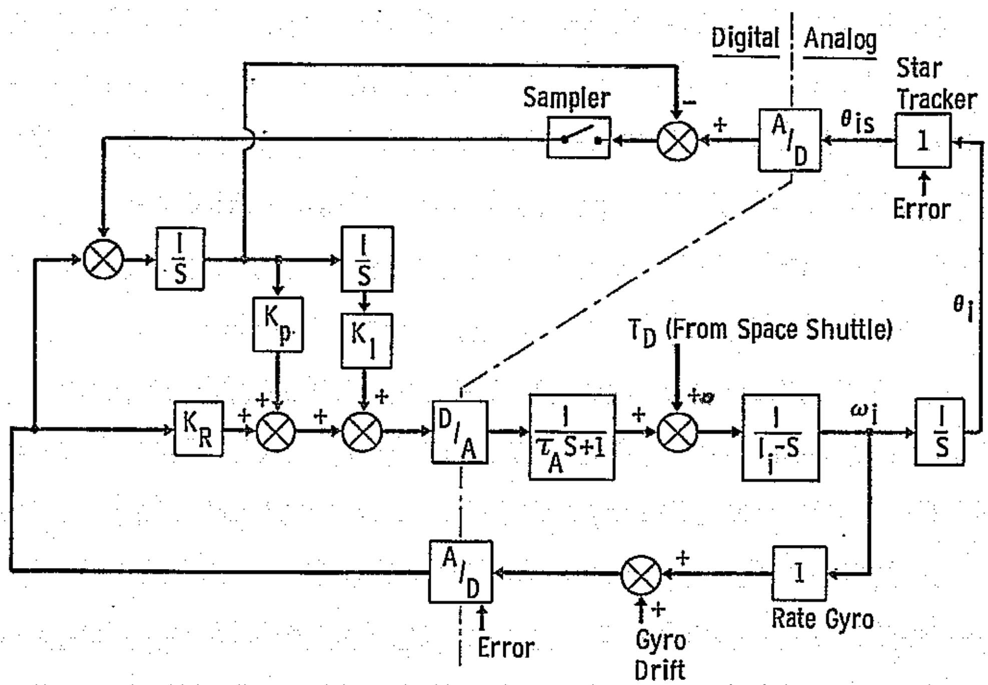


Figure 5.5.5-1 Static Error Budget System Block Diagram

For the purpose of this study two types of rate gyros (64 PMRIG and Gyroflex) and two levels of star tracker calibrated accuracy (1 arc sec, 10 arc sec) were considered. Time between star tracker

updates, and quantization level magnitudes of the star tracker and rate gyro signals are treated as variables.

Discussion - An equation expressing the instrument static attitude error in any of the three axes as a function of its error sources was derived from Figure 5.5.5-1 in the following manner.

It is assumed that in steady state no torque is acting on the vehicle. This implies that the disturbance torque, T_D, is being exactly cancelled by a constant output from the integrator leading to the integral gain, K_I. Contributions from the other two control law paths via K_P and K_R are both identically zero. This implies that the rate gyro signal on the digital side is also identically zero. In this manner the rate signal, ω_i, which is in error due to rate gyro signal quantization and rate gyro drift is integrated to yield attitude. The attitude error obtained in this fashion is additive to the error already existing in the attitude signal due to star tracker calibration error and star tracker signal quantization. The attitude error due to star tracker calibration offset and signal quantization exists even after star tracker updates. The static attitude error equation is thus given by:

$$X_\theta = X_{ST} + X_{STQ} + T\ (X_{RG} + X_{RGQ})$$

Note that the contribution to X_θ due to the rate gyro errors is a function of time, T, since the last star tracker update. The variables are defined as follows (where X denotes a random variable):

X_{ST} - error due to star tracker calibration offset

X_{STQ} - error due to star tracker signal quantization

X_{RG} - error due to rate gyro drift

X_{RGQ} - error due to rate signal quantization

By applying the expected value function to the expression of X_θ it is evident that $E(X_\theta) = 0$ since the expected (average, mean) values of X_{ST}, X_{STQ}, X_{RG}, and X_{RGQ} are all zero. It is thus useful to determine the variance of X_θ.

Assuming all of the error random variables to be independent, the variance of X_θ can be expressed as:

$$V(X_\theta) = V(X_{ST}) + V(X_{STQ}) + V(T \cdot X_{RG}) + V(T \cdot X_{RGQ})$$

It is assumed at this point that the random variables X_{ST} and X_{RG} are normally (Gaussian) distributed with mean $\mu = 0$ and variances σ^2_{ST} and σ^2_{RG}, respectively. The random variable X_{STQ} and X_{RGQ} are uniformly distributed with mean $\mu = 0$ and variance $V(X_{STQ})$ and $V(X_{RGQ})$, respectively. The errors X_{STQ} and X_{RGQ} are assumed uniformly

distributed between plus one-half and minus one-half their respective quantization levels Q_{ST} and Q_{RG}. It can be shown on the basis of assuming the uniform distribution that:

$$V(X_{STQ}) = \frac{Q_{ST}^2}{12}$$

and

$$V(X_{RGQ}) = \frac{Q_{RG}^2}{12}$$

Also, it follows that:

$$V(T \cdot X_{RG}) = T^2 \, V(X_{RG})$$

and

$$V(T \cdot X_{RGQ}) = T^2 \, V(X_{RGQ})$$

Finally $V(X_\theta)$ can be expressed as follows:

$$V(X_\theta) = V(X_{ST}) + \frac{Q_{ST}^2}{12} + T^2 \left[V(X_{RG}) + \frac{Q_{RG}^2}{12} \right]$$

Results - Calculations were performed to determine how the standard derivation σ_θ of the static pointing error (where $\sigma_\theta = \sqrt{VAR(X_\theta)}$) increases with time, T, between star tracker updates. The following four separate cases of rate gyro/star tracker combinations were considered:

(a) Gyroflex rate gyro ($\sigma_{RG} = 0.02$ arc-s/sec) and star tracker ($\sigma_{ST} = 10$ arc sec)

(b) Gyroflex rate gyro ($\sigma_{RG} = 0.02$ arc-s/sec) and star tracker ($\sigma_{ST} = 1$ arc sec)

(c) 64 PMRIG rate gyro ($\sigma_{RG} = 2.5 \times 10^{-4}$ arc-s/sec) and star tracker ($\sigma_{ST} = 10$ arc sec)

(d) 64 PMRIG rate gyro ($\sigma_{RG} = 2.5 \times 10^{-4}$ arc-s/sec) and star tracker ($\sigma_{ST} = 1$ arc sec)

For each of these four cases three different quantization levels of the rate gyro and star tracker signals were investigated. A nominal set of quantization levels for each case was obtained by equating the contribution to static errors of the star tracker quantization and star tracker accuracy as well as rate gyro signal quantization and rate gyro drift. That is, it is assumed that the nominal quantization levels can be designated such that $\sigma_{ST}^2 = Q_{ST}^2/12$ and $\sigma_{RG}^2 = Q_{RG}^2/12$.

Two additional quantization levels investigated were 10 x nominal and 1/10 x nominal of the respective star tracker and rate gyro signals. The results for cases (a) through (d) are summarized in graphical form in Figures 5.5.5-2 through 5.5.5-5. In these groups the standard deviation of the static error is plotted against time, T, between star tracker updates for each of three quantization levels.

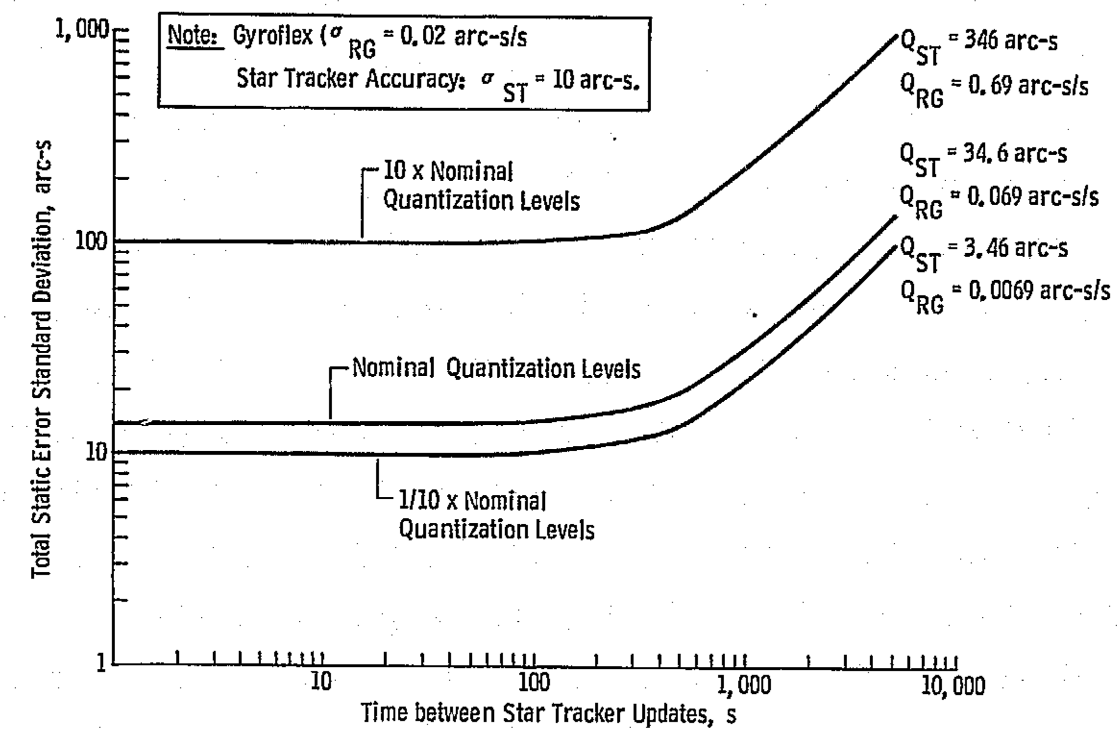


Figure 5.5.5-2 Static Error Budget Estimate - A

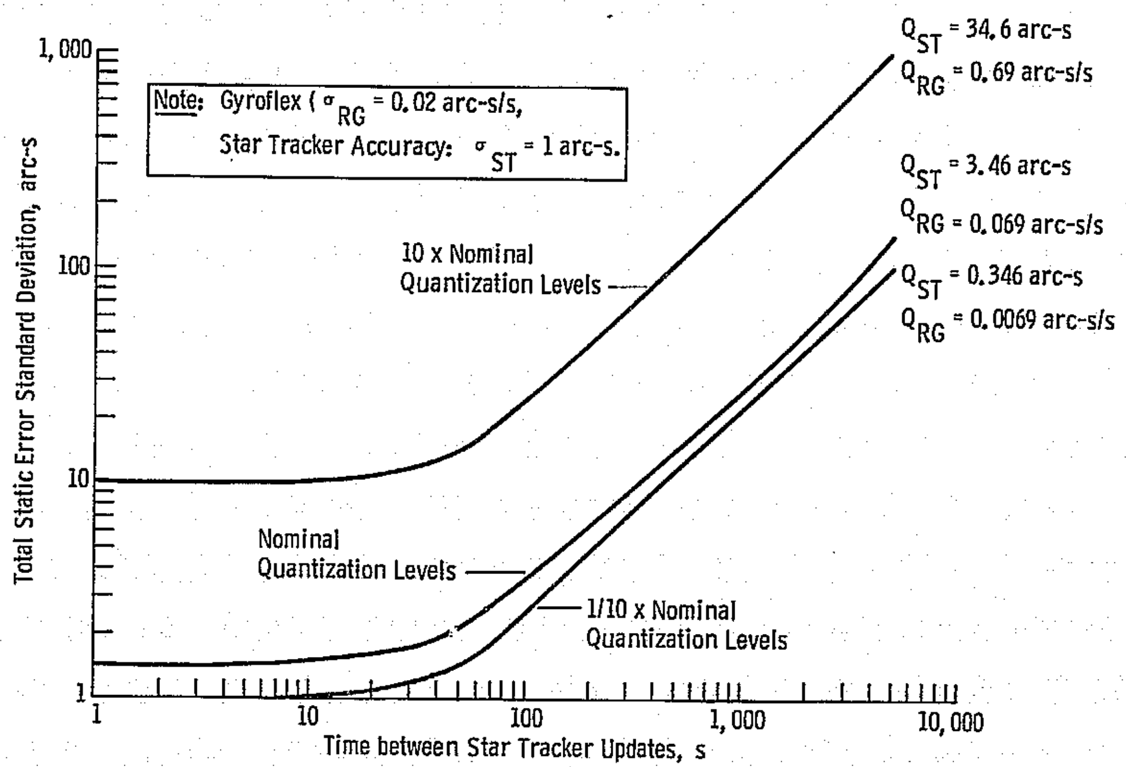


Figure 5.5.5-3 Static Error Budget Estimate - B

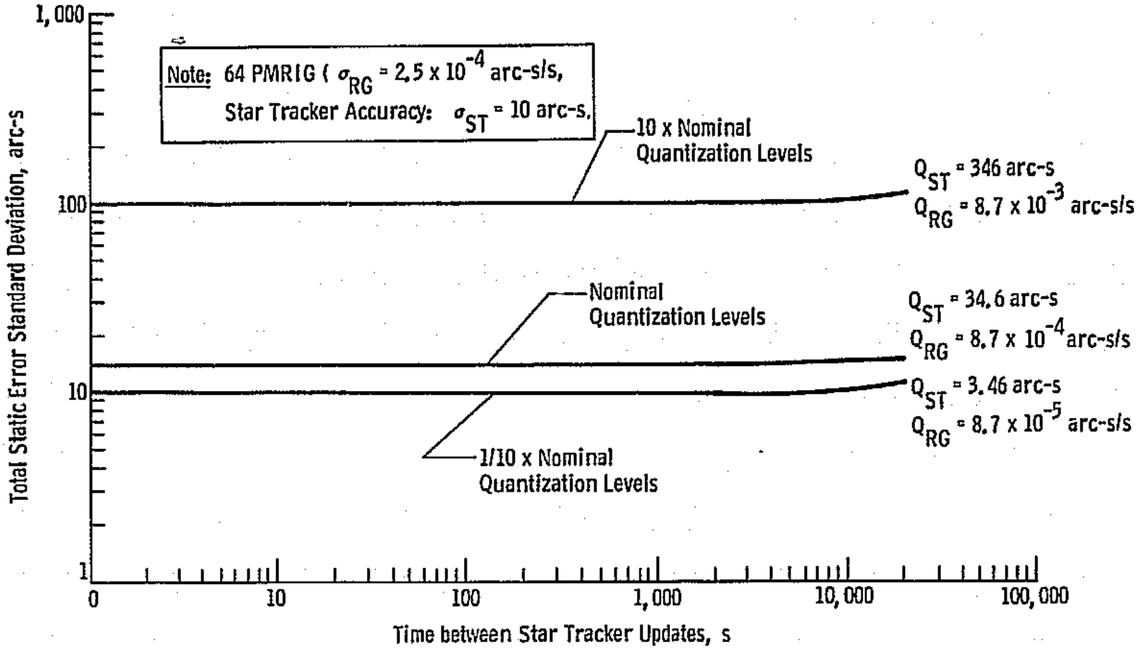


Figure 5.5.5-4 Static Error Budget Estimate - C

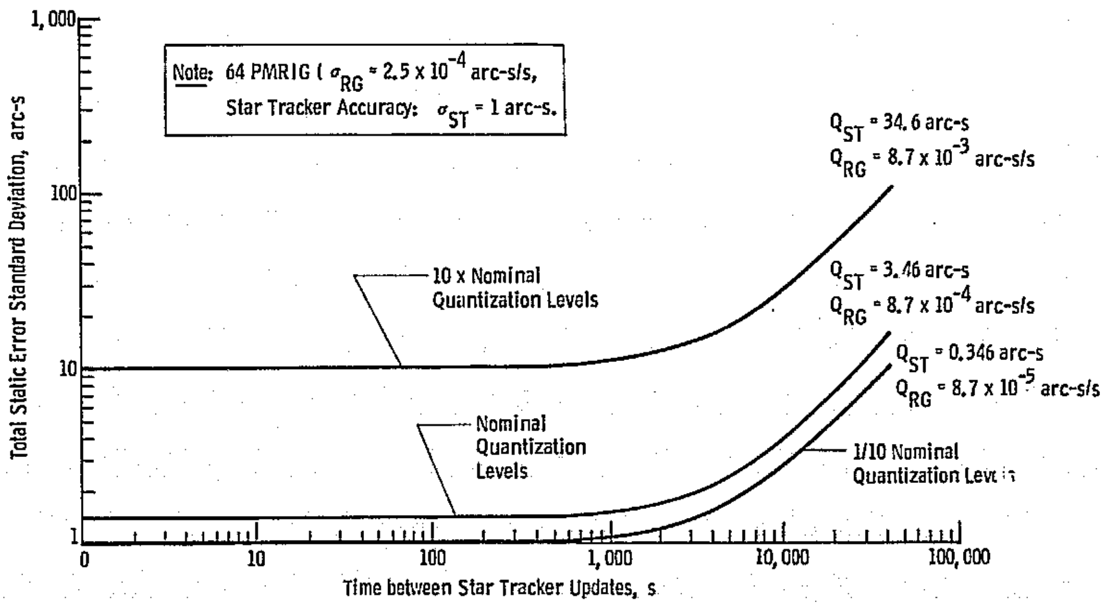


Figure 5.5.5-5 Static Error Budget Estimate - D

5.5.6 Attitude Reference System

Instrument attitude control is based on the alignment of the instrument coordinate system to a reference coordinate system, which can be either inertially fixed or moving, in the form of the quaternion Q_{IR} describing the instrument attitude relative to the reference attitude (see Figure 5.5.6-1). The reference attitude is established by the instrument rate command ω_{IC}, and instrument attitude is obtained by periodically updating rate gyro calculated instrument attitude with star tracker measured instrument attitude.

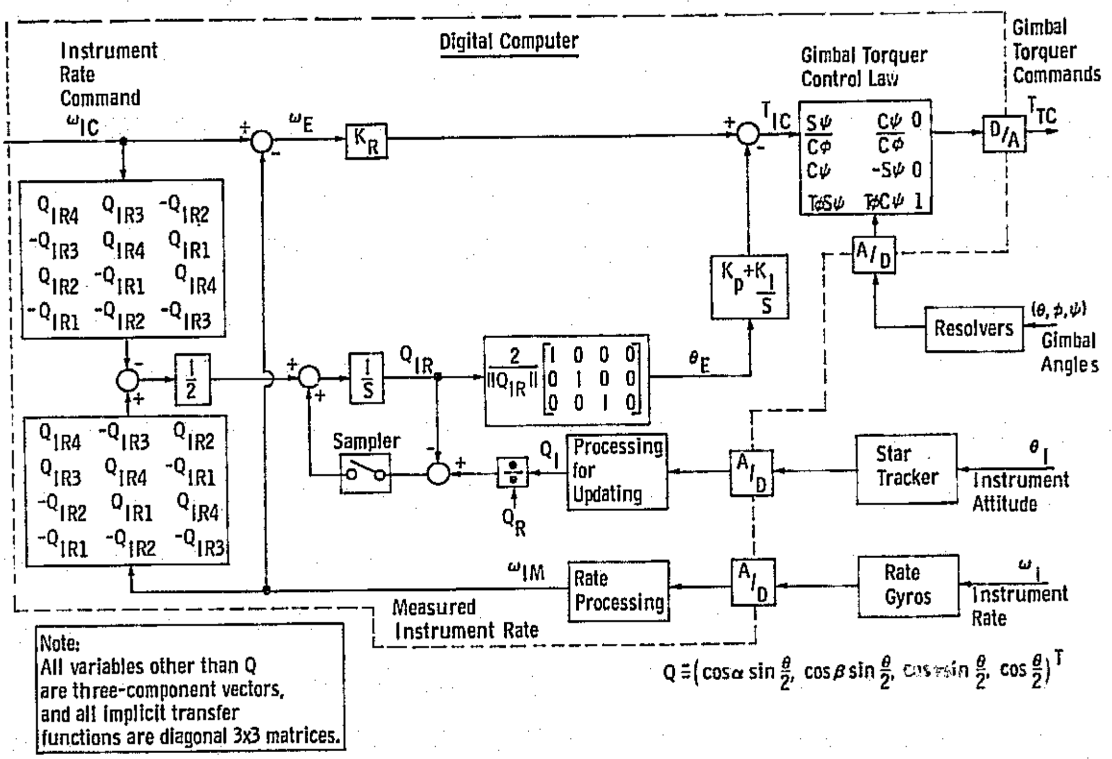


Figure 5.5.6-1 Attitude Reference System

A fine pointing control loop requires a wide bandwidth attitude sensor. However to have a high probability of having a useable star in its field of view, a star tracker must look at dim stars. The incoming power of a dim star is not sufficient to allow a wide bandwidth. Rate gyros have wide bandwidth but when used as attitude sensors are inaccurate due to rate gyro drift. The solution is to combine the two by periodically updating rate gyro strapdown attitude computations with star tracker measured attitude.

As long as ω_{IC} is constrained to values that the instrument can realize through its attitude control system, the attitude errors will

remain small even though large angle maneuvers are performed. Hence instrument attitude error θ_E is approximated as two times the first three components of Q_{IR} normalized to a magnitude of one to reduce integration errors. Instrument control torque command T_{IC} is a linear combination of instrument rate error ω_E (for stability), instrument attitude error and the integral of instrument attitude error (for accuracy). The Gimbal Torquer Control Law resolves T_{IC}, which is in instrument space, into specific gimbal torquer commands T_{TC} for the outer, middle and inner gimbals.

5.5.7 Orientation Updates With FHST Close To Earth's Limb

Viewing Restrictions from Physical Obstructions - To avoid occultation of selected guide stars, the FHST must view the narrow strip between the spacecraft structures and the earth's limb, if the Orbiter is in the Z-LV mode. This requires that the FHST axis must be offset from the axes of the limb-viewing instruments. A sufficient offset will enable the FHST to acquire and track threshold stars effectively, if background effects from unwanted earthshine (albedo), spacecraft scattering, and sunshine can be properly controlled.

From an orbital altitude of 209 km as specified for the early AMPS missions, all the significant atmospheric refraction effects are limited to the band within 4 degrees from the limb. The rays that come from stars that appear 4 degrees above the limb dip down to a minimum atmospheric height of 100 km (tangency point). The first constraint that arises is that the FHST field-of-view should not be so low as to pick up stars that are disturbed by the lower atmosphere. Since the FHST field-of-view is defined as 8 x 8 degrees, the axis of the FHST should not be closer than 8 degrees from the limb.

The viewing limitations in the upward direction, with the Orbiter in the Z-LV attitude, are defined by the thermal radiators. Based on scaling of available Orbiter drawings, the geometry shown in Figure 5.5.7-1 was determined.

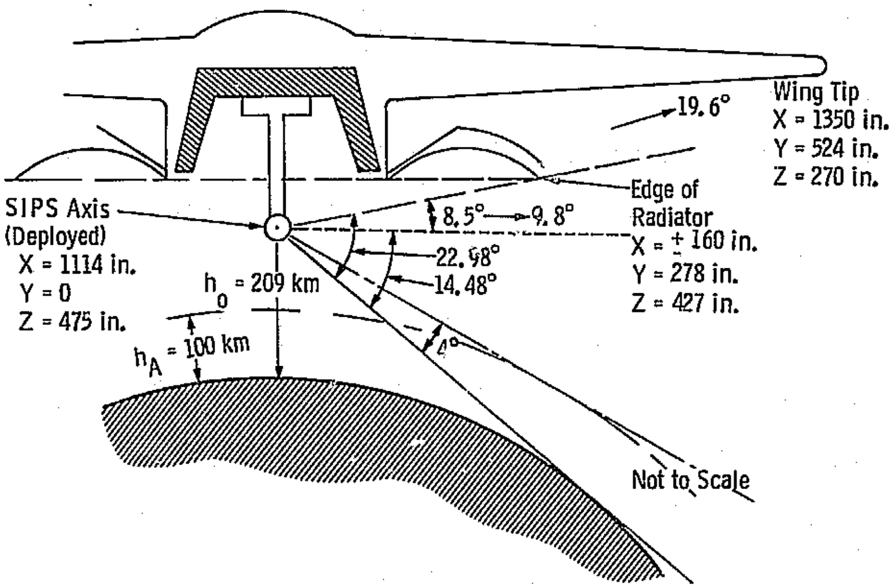


Figure 5.5.7-1 Geometry of Star Tracker Viewing Restrictions

The worst-case (narrowest strip) geometry is obtained when observing forward or aft of the Orbiter Y axis by 40 degrees, at which point the radiator outer edge is at 81.5 degrees from the zenith, and the earth's limb at 75.52 degrees (ignoring atmospheric refraction effects) from the nadir so that the separation between the two is only 22.98 degrees.

Scattering Effects and Restrictions - The background effects due to earthshine albedo, spacecraft structures, and direct solar radiation present more stringent demands than merely allowing clear viewing of the stars. To avoid performance deterioration, based on the data obtained from a telecon with Ball Brothers Research Corp. (BBRC), a two-stage sunshade is adequate if the sun does not reach the inner stage. Similarly, if no direct albedo is allowed to reach the front lens of the FHST objective, acquisition and track performance on limiting-magnitude stars should not be impaired.

Both of these background sources, as well as the scattering from the Orbiter surfaces, are controlled by the sunshade. A minimum-width sunshade that provides the required off-axis attenuation of unwanted light is shown in Figure 5.5.7-2. By orienting the FHST axis so that it is aimed 11 plus degrees above the limb when the Orbiter is in the Z-LV mode, the albedo and spacecraft reflections are controlled so that the predicted minimum-brightness stars (visual magnitude +5.7) can be acquired and tracked.

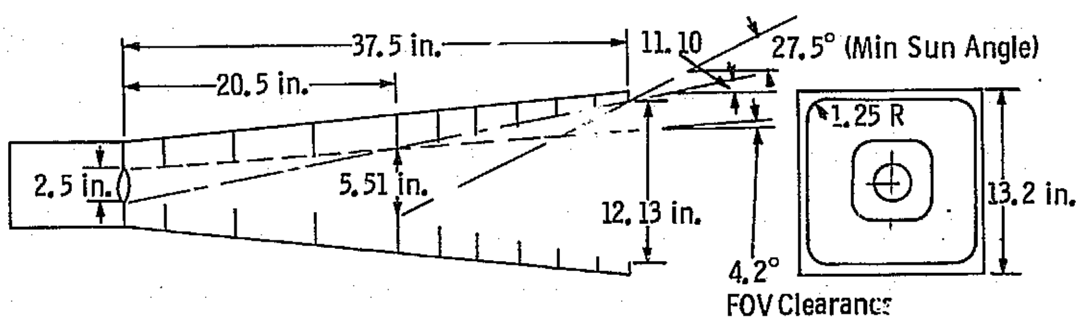


Figure 5.5.7-2 Optical Diagram for FHST Sunshade for AMPS

Note that the inner stage of the sunshade is not directly sunlit for all sun angles greater than 27.5 degrees. This restriction can be met even if the Orbiter is maintained in the X-POP, Z-LV attitude and the angle between the sun vector and the orbital plane (β-angle) is small by offsetting the viewing direction relative to the Orbiter's Y-axis.

Star Availability - To assess the probability that an adequate star will be available within the field of view of the FHST at the time that an update is needed, a simple analysis of probability was performed.

The data on star densities was obtained from Allen's Astrophysical Quantities," Athalone Press (second edition) 1963. This reference describes the variation of star number density per square degree with galactic latitude; a significant variation is noted from galactic equator to galactic poles. The Allen data is detailed for "photographic magnitudes" only, and in increments of a full magnitude. To convert and interpolate, several factors were used, derived from analysis of the tabulated data:

- ratio of mean star density to density at the galactic poles, all stars brighter than a specified photographic magnitude: 1.89:1
- ratio of star densities for $m_v = +5.7$ compared to $m_v = +5.0$: 2.15:1

  (ratio of star densities brighter than a specified visual magnitude to star densities for an equal numerical value of photographic magnitude: 1.66:1)
- mean star density for stars $m_v \leq +5.0$: 0.040 stars per square degree.

The anticipated density at the galactic poles of guide stars adequate for tracking is therefore,

$$N_m(b^{II} = \pm 90^o,\ m_v \leq +5.7) = \frac{0.040 \times 2.15}{1.89} = 0.045 \text{ stars/square degree}$$

and similarly,

$$N_m(\text{mean},\ m_v \leq +5.7) = 0.085 \text{ stars/square degree}$$

$$N_m(b^{II} = 0^o,\ m_v \leq +5.7) = 0.184 \text{ stars/square degree}$$

For a worst-case analysis, the star density at the galactic poles should be used, even if the orbit parameters and the spacecraft attitude constraints should make it impossible to orient the FHST toward the galactic poles. The distribution of stars in any given region is reasonably uniform, so that the probability of encountering a suitable star within a solid angle may be considered to be approximately the solid angle times the star density per unit solid angle. Therefore, if a region of 22 square degrees (4.7 x 4.7 degrees) is searched, the probability encountering at least one adequate guide star should approach unity. Whereas the probability of encountering a star in two adjacent unit solid angle intervals should be independent, some estimate of the true probability can be obtained by the rationale that the probability of not finding a star in 64 square degrees should be the probability of not finding the star in the first square degree to the 64th power:

$$\overline{P}\ (m_v \le +5.7,\ 64\ \text{sq deg}) = (\overline{P}(m_v \le +5.7;\ 1.0\ \text{sq deg})^{64}$$

$$= (1-P\ (m_v \le 5.7;\ 1\ \text{sq deg})^{64} = (1 - N_m{}^{v}(b^{II} = 90^\circ;\ m_v \le +5.7))^{64}$$

$$= (0.955)^{64} = .05$$

This certainly should be interpreted as a worst-case situation, since probabilities for different intervals are not dependent on each other. But it does indicate that since no performance degradation should be expected in the FHST if the correct sunshade design is used, the probability of acquiring an adequate star at any instant should be 0.95 or greater, i.e., approaching unity.

Conclusions - By installing the FHST in the same SIPS canister as a limb-viewing instrument but oriented so that its axis is 11.5 degrees above the limb, and adding the proper sunshade, satisfactory performance of the FHST for orientation updates can be obtained with the Orbiter in the Z-LV attitude.

5.6 Data Management Subsystem

5.6.1 Data Management/Controls and Display Interface Analysis

The AMPS program as envisioned for the 1980's will require quick turnaround capability from flight-to-flight. In addition, a low cost approach requires that support equipment should be capable of handling a variety of experiment requirements. In terms of the data management subsystem, the rack mounted signal conditioner and processing equipment located inside the Spacelab Module represent an area where the previous criteria may be applied. In particular, a modular approach would enhance quick turnaround, minimum checkout time and cost reduction by spreading the hardware cost over many flights. The feasibility of providing these features and in particular the use of NIM/CAMAC hardware is investigated.

5.6.1.1 Requirements

The Phase B study has indicated a need to provide mission dependent hardware at the control and display station of the Spacelab module. Inherent in the design of such equipment is the need for quick changeout, ease of checkout and applicability over a broad spectrum of payloads and missions.

For Flights 1 and 2, digital, analog and video signals must be processed or rerouted at the control and display station. These requirements are detailed as follows:

Flight 1

(1) OBIPS - Two video signals are routed to the Spacelab module for display, recording or downlink transmission. One video signal originates from the aft pallet and the second video signal is from the OBIPS on the RMS. A dedicated video recorder and monitor are required. Input switch and recorder switching are required. See Figure 5.6.1-1.

(2) Electron Beam Diagnostic (Analog) - The electron beam diagnostic instrument (mounted on the RMS) provides 7 analog signals which are frequency multiplexed with digital data and transmitted to its pallet mounted receivers. This multiplexed signal must be discriminated whereby the digital signal is routed to an RAU input and the composite analog signal is further demultiplexed to reconstruct the original analog signal. (See Figure 5.6.1-2.) This signal is then available for onboard viewing. The multiplexed analog signal is also routed to an onboard display and to either a tape recorder or directly to the Orbiter for real time transmission.

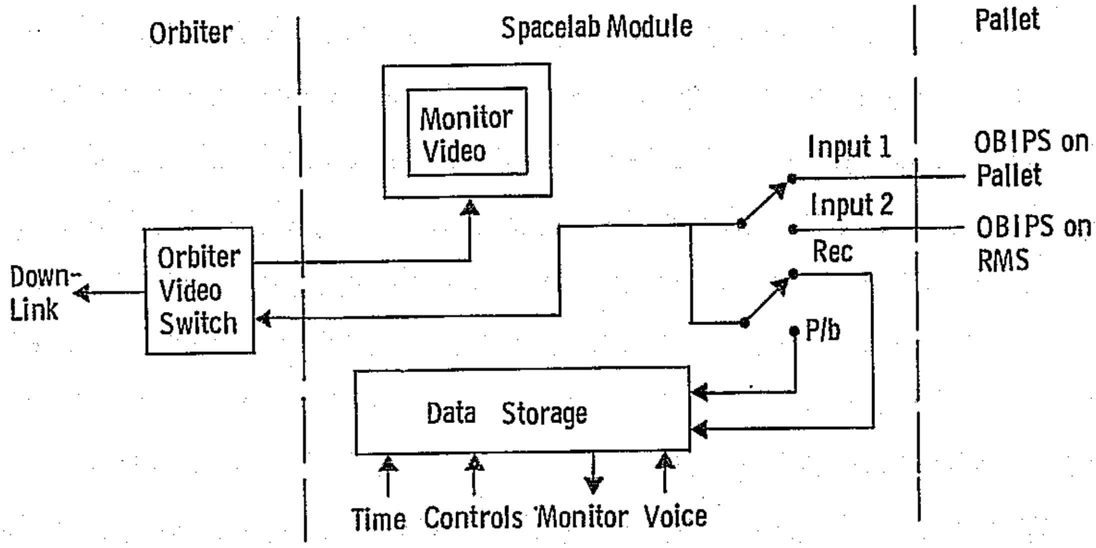


Figure 5.6.1-1 OBIPS Television Requirements (Flight 1)

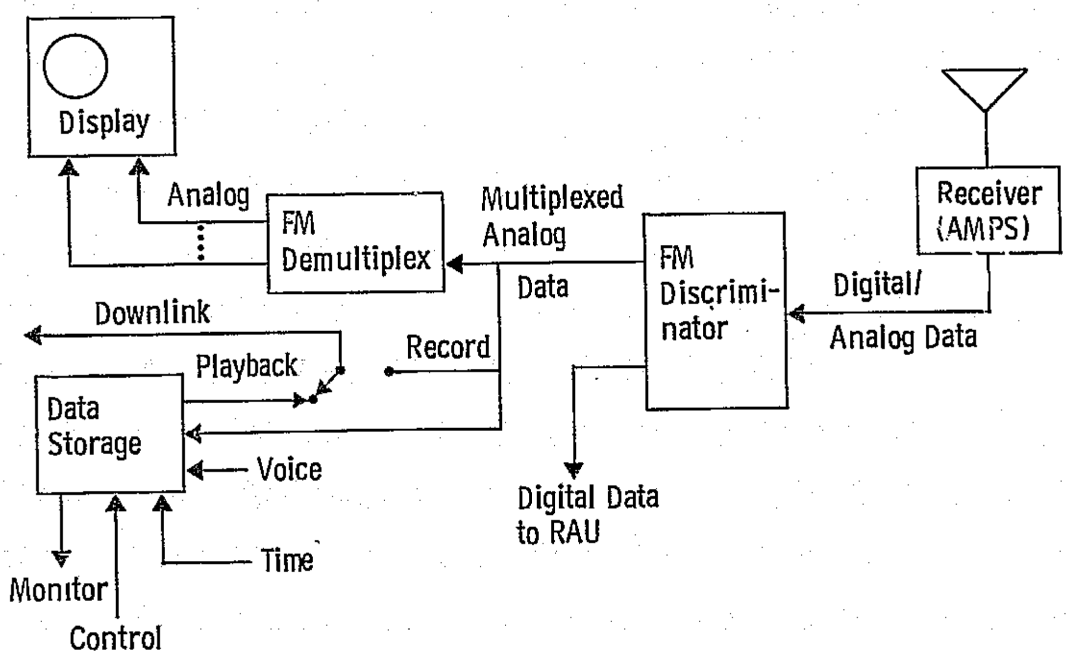


Figure 5.6.1-2 Beam Diagnostic Analog Requirements (Flight 1)

(3) Electron Accelerator (Analog) - Pulse signals from the electron accelerator must be provided for onboard display and retrieved for post flight analysis. (See Figure 5.6.1-3.) A dedicated recorder is required to record typical signal outputs at short time intervals. Selected outputs will be downlinked for ground monitor.

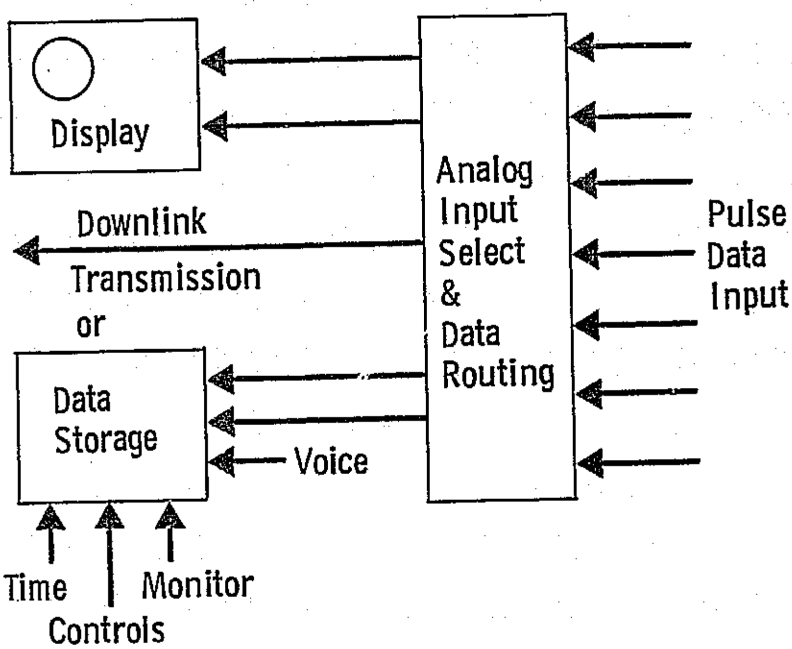


Figure 5.6.1-3 Electron Accelerator Pulse Data Requirements (Flight 1)

Flight 2

(1) RF Receiver Package - A 30 KHz IF signal is multiplexed with 4 Kbps of digital data and transmitted from the deployed package to the AMPS provided receiver. (See Figure 5.6.1-4.) The received signal is routed a) to an FM discriminator for onboard display and monitor and b) to dedicated subcarrier oscillators where the received signal and a 30 KHz IF signal originating from instruments mounted on the pallet are combined for recording or downlink transmission.

(2) Plasma Wake Diagnostic - Analog and digital signals from the RMS mounted diagnostic package are multiplexed and transmitted to the AMPS pallet receiver. The received signal is demodulated and the digital data routed to an RAU. (See Figure 5.6.1-5.) The analog signals are simultaneously routed to a) FM discriminators where the three analog channels are restored for onboard display and b) to either a dedicated recorder or the Orbiter Ku-band for real time transmission.

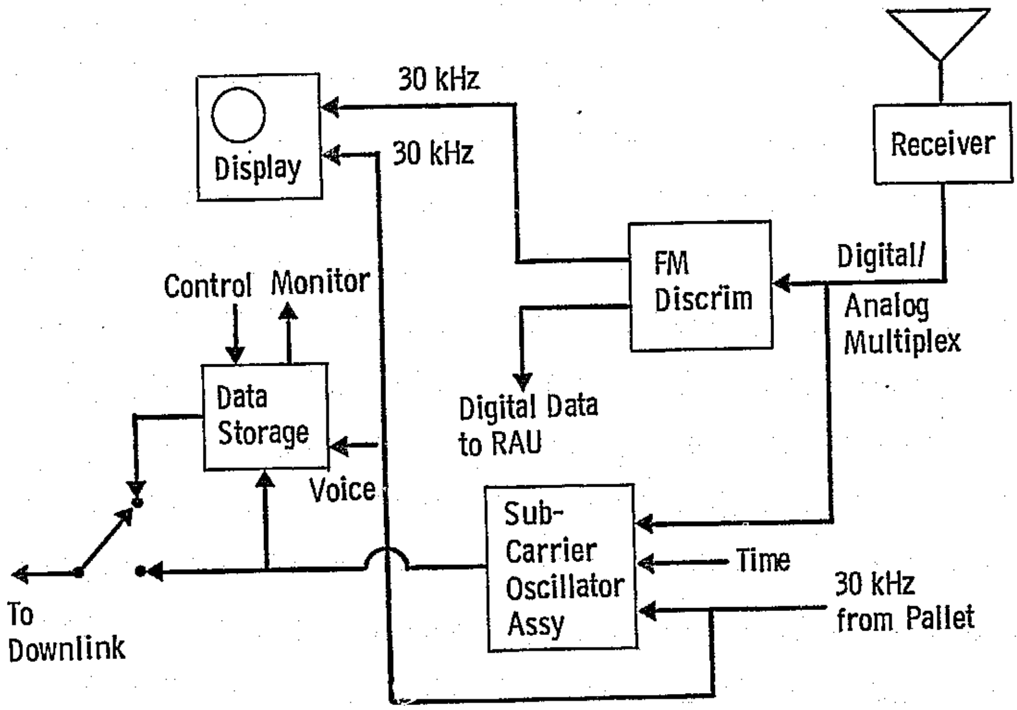


Figure 5.6.1-4 RF Receiver Package Requirements (Flight 2)

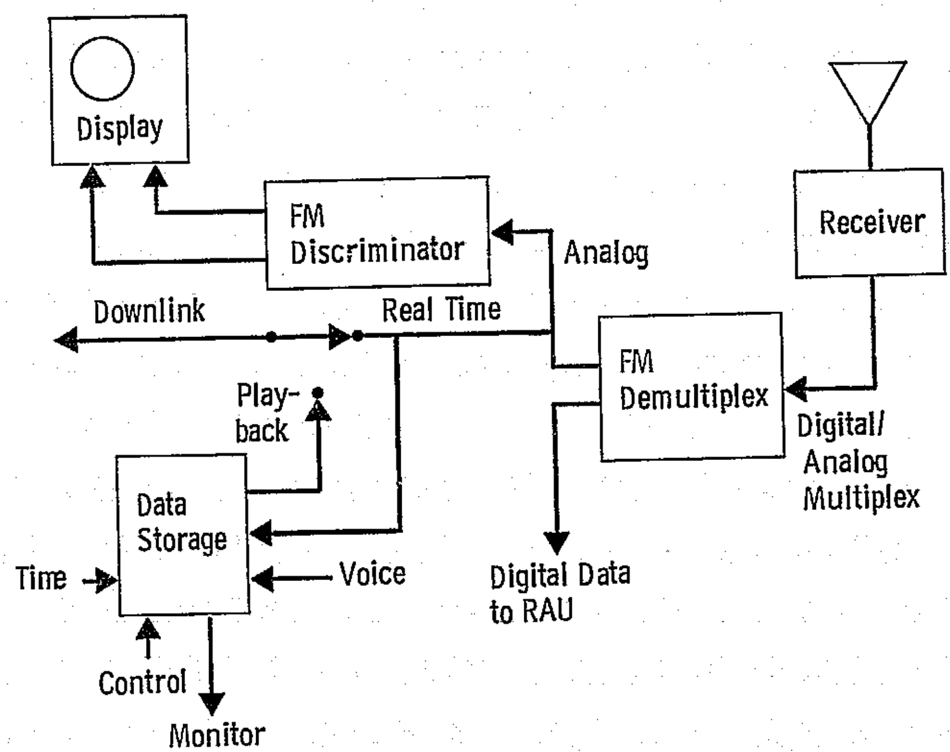


Figure 5.6.1-5 Plasma Wake Diagnostic Package (Flight 2)

5.6.1.2 Discussion

The requirements identified in the previous section are summarized in Table 5.6.1-1. The table identifies the primary need for data storage and onboard display of various signals. A digital interface with the Spacelab RAU is required as well as FM modules for three experiments. Support functions such as controls from an RAU and GMT are also noted.

Table 5.6.1-1 Control and Display Interface Requirement Summary

Parameter \ Experiment	OBIPS (TV)	Beam Diagnostic	Electron Accel.	RF Receiver Package	Plasma Wake Diagnostic	Future Needs: Dedicated Panel
TV Monitor	X					
TV Switch	X					
Video Recorder	X					
FM Discriminator		X		X	X	
Data Storage		X	X	X	X	
Oscilloscope I/P Switching		X	X	X	X	
Digital Data to RAU		X		X	X	
Signal Conditioning			X			
FM Subcarrier Oscillator Assembly				X		
Plotter/Driver						X
Digital Modules						X
Voice						
Time	X	X	X		X	X
Controls From RAU	X	X	X	X	X	X
Monitor Functions to RAU		X				X

The key point in implementing a system to meet these requirements are standardized modules and flexibility of routing data, controls and timing signals. A functional schematic incorporating the requirements previously discussed is presented in Figure 5.6.1-6. Four signal routing/switching panels are provided for the necessary flexibility to meet mission to mission requirements. These panels, identified as (1), (2),(3) and (4) are configured as shown prior to a mission and modified during a flight only if a work-around or contingency occurs.

All input signals are routed via panel (1) to FM modules, dedicated support equipment, throughput to panel (2) or to a video/analog switching panel (4). Signal switching panel (2) provides the interface between demultiplexed analog signals, pulse stretched signals, throughput signals, and oscilloscopes, recorders or plotters. Panel (2) is shown in the figure as a functional block whose actual implementation would be accomplished by manual or automatic switching.

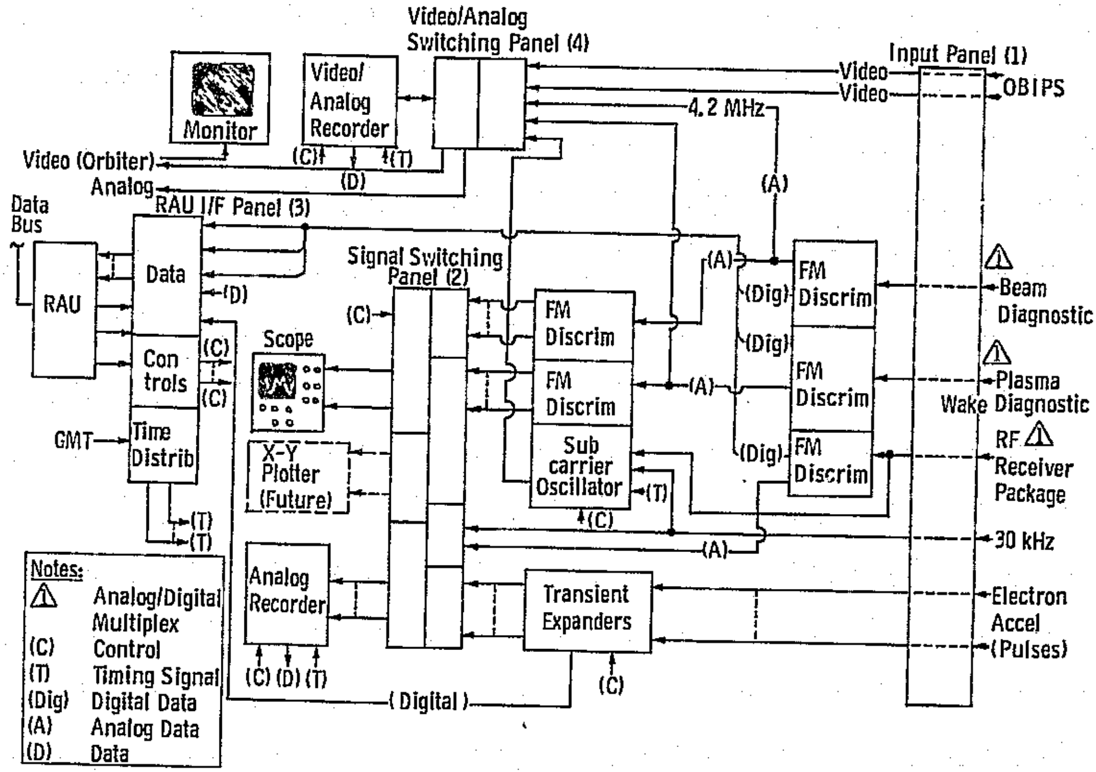


Figure 5.6.1-6 Flight 1/2 Control and Display Interface

Panel (3) is provided as a means of interfacing data and controls with an RAU. This panel would incorporate the necessary equipment for data buffering, isolation, and amplification. Digital and housekeeping data are routed to the RAU. Control functions are distributed from this panel primarily to the recorders. Other control functions would include band-edge calibration of the subcarrier oscillators. This panel would also include capability for time distribution and would supplement the time distribution capability provided by the Spacelab Remote Amplifier and Advisory Box.

The video/analog switching panel (4) routes the input signals to either the recorder or to the Orbiter. Analog inputs are FM multiplexed data ranging in bandwidth from a few thousand hertz to 4.2 megahertz. Signals to the video monitor are received from the Orbiter video switch.

The flexibility provided by this system allows for easy installation and minimal impact during flight reconfiguration. Signals can be routed from panel to dedicated equipment or panel-to-panel. Dedicated equipment such as the transient expanders between panels (1) and (2) are shown. This unit, which has a digital output, is routed to the RAU interface panel (3) as an alternate means of data acquisition. The FM discriminator outputs are routed to the video/analog switching panel (4), RAU interface panel (3) or to the signal switching panel (2). A

modular and flexible system such as this can take advantage of the on-board crew in terms of replacing failed components or possibly reconfiguring the system. For the most part built-in redundancy is not required. The system is capable of accommodating dedicated panels as well as various displays. An X-Y plotter is shown as a future installation.

5.6.1.3 NIM/CAMAC Application

In line with the AMPS concept of developing an experiment by use of instruments obtained from a common inventory, establishment of an inventory of flight support equipment, especially in the interface between data management and controls and display can provide similar benefits. While cost savings will be a function of equipment applicability to a broad spectrum of payloads, a modular approach to hardware application can provide benefits in the area of quick changeover for mission turnaround, ease of checkout and replacement of failed modules in-flight instead of designing built-in redundancy. This approach requires a standardized interface whereby many modules with different functions will perform its intended function when inserted into a common housing.

The use of the modular concept and standardization as applied to the nuclear industry has attracted close scrutiny for Shuttle payload application. This system, commonly referred to as the NIM/CAMAC system (Nuclear Instrument Module/Computer Automated Measurement and Controls), is in wide use in ground laboratories here and abroad. The applicability of this modular system for the AMPS control and display interface is discussed.

Background - NIM-CAMAC standards were developed for ground-based laboratory nuclear instrumentation equipment in order to reduce the need for one-of-a-kind electronics that made nuclear instrumentation a high-cost item for experiments. Use of NIM-CAMAC standards has been extended to astronomy, medical electronics, and industrial process control. There are many manufacturers in the United States and Europe who manufacture electronic modules to NIM-CAMAC standards. Demand for the modules is such that competition has resulted in making the modules economical and reliable, and has assured a continued development of new modules to expand the electronic functions available to an experimenter.

NIM standards were written by the United States AEC Committee on Nuclear Instrument Modules to assure mechanical and electrical interchangeability of transistorized modular instruments. The standards do not specify data acquisition systems. NIM modules are primarily analog and are housed in a standard "bin" with a power supply. Representative NIM modules include various type amplifiers, scalers, timers, discriminators and power supplies.

CAMAC modular instrumentation system standards for data handling were initiated by the ESONE Committee of European Laboratories in cooperation with the United States AEC NIM Committee, who subsequently endorsed them. The standards specify a system suitable for digital data acquisition (Dataways) using a computer.

The attractiveness of this system is that a) NIM/CAMAC is governed by an agreed upon standard and b) its standardized modular hardware is envisioned as the answer to low cost payloads.

An inventory of different modules are available which are inserted into the CRATE as required. Each "CRATE" also has a controller module which, as applied to Shuttle, will probably be a micro-processor. The system is capable of operating in two modes; either driven from the system computer or independently from the controller within the "CRATE." For example implementing a sequence of operation could be done by the controller while data processing would be done by the computer.

Application to Controls and Displays Interface - The primary application of NIM/CAMAC for the subject interface is standardization of AMPS analog modules to NIM requirements and the use of CAMAC modules to enhance digital data processing. Figure 6.5.1-7 shows the use of NIM bins and CAMAC modules for AMPS Flight 1 and 2 plus expansion capabilities to support other payloads. All FM multiplex/demultiplex equipment is housed in NIM bins. Fan-out modules are provided to route the same signal to multiple destinations. The modules themselves would be FM discriminators and subcarrier oscillators which would be new build items. The NIM bin could also be used to house video modules such as line drivers, isolation amplifiers, etc.

CAMAC hardware can be used to enhance the flexibility at the control and display interface as noted in the same figure. A CAMAC crate with a controller would interface with an RAU for bidirectional communication with the experiment computer. CAMAC modules would be used as a data interface between a data source and the experiment computer. Analog data could be multiplexed and A/D converted while digital data would interface with input registers. Digital data from the computer could be routed through a D/A converter for on-board display. Other applications include digitizing analog data, storing it in memory and providing later recall by D/A conversion. Occasions may arise whereby it is necessary to monitor both the oscilloscope for analog data and alphanumeric data on a separate CRT display. CAMAC character generator and display driver modules could be used to provide alpha numeric data and analog data on a single oscilloscope.

The use of the CAMAC crate deletes the need for the RAU interface panel of Figure 5.6.1-6 since most of these functions are handled via CAMAC modules. Although there is some redundancy between CAMAC modules and an RAU, the flexibility available with the use of standardized modules provides a mechanism which fits the requirements for quick checkout and fast turnaround capability between flights. Implementation

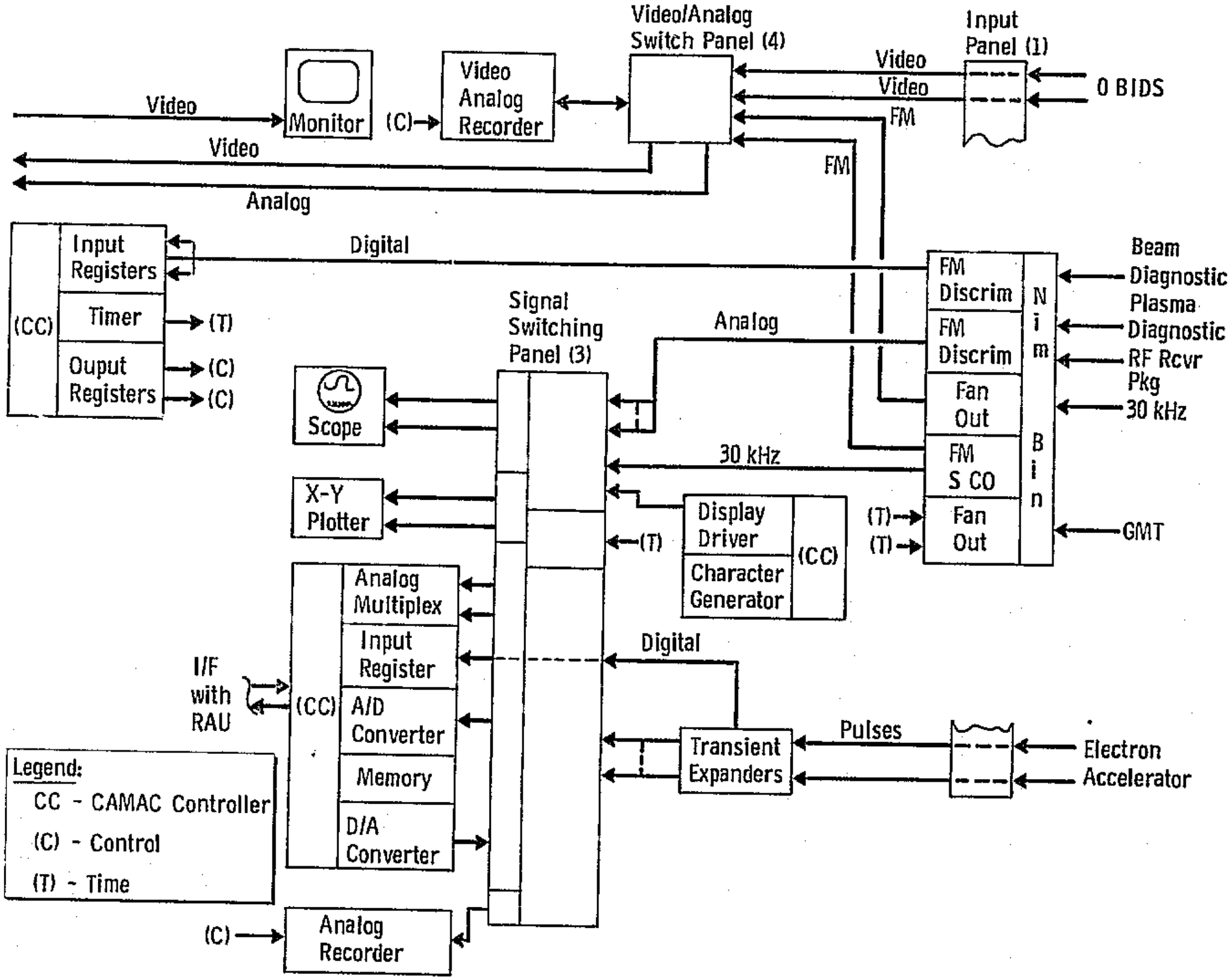


Figure 6.5.1-7 NIM/CAMAC Application to Control and Display Interface

of NIM/CAMAC hardware for flight usage is not without redesign problems. On-going studies and tests of CAMAC modules indicate that with modifications such modules can be used for Shuttle application. Modification in the area of power reduction, mechanical design, connectors and environment are being evaluated by various NASA centers. In addition, the interface between the CAMAC crates and the computer should be modified from the present parallel system to a serial interface. The application of CAMAC to AMPS assumes satisfactory solutions to the problems previously mentioned.

Conclusions - Because of the austere nature of the Spacelab module to accommodate AMPS analog/video signals, dedicated equipment is required which should be designed with flexible signal routing capability. The use of signal switching and interface panels can provide for use of this type hardware across a broad spectrum of AMPS flights as well as other payloads. The NIM bin represents a standardized reference for analog module design while CAMAC modules can enhance signal data processing for onboard display. Existing module design can be used effectively to provide a number of services to support the onboard crew in the area of experiment management.

5.6.2 TM Format and Data Correlation

With the multiplicity of data sources and data rates inherent in the AMPS payload, efficient ground data recovery will depend on a uniform set of guidelines and telemetry formats with proper overhead data to provide unambiguous data codes. Telemetry format requirements are best established by reviewing an information system in reverse flow of data, i.e., postflight data analysis and its associated problems which, in many cases, are solved by incorporating the desired solutions into the telemetry format. This includes such parameters as, ease of data decommutation, data correlation, data time skew and standardized formats. In the AMPS program this problem is compounded by the fact that supporting data such as the vehicle attitude state vector and remote manipulating arm position data are generated by the separate Orbiter computer/data system. Correlation of Orbiter data with the AMPS payload data will depend on the usage of a common time base and knowledge of the time delays associated when the data was sampled and when it was inserted in the telemetry data stream.

5.6.2.1 Requirements

The major requirements for the AMPS payload digital format are identified in the following documents:

Aerospace Data System (ADS) Standards, No. X-560-63-2, Goddard Space Flight Center, Greenbelt, Maryland

Shuttle Telemetry Data Format Control Book (Preliminary), Dec 1975, Johnson Space Center, Houston, Texas

The requirements identified in these documents are tabulated in Table 5.6.2-1. This applies only to those data that are not interleaved with the Orbiter data but are transmitted to the ground as an independent data stream.

Waivers of the ADS standards are permitted contingent upon approval of Data Systems Requirements Committee (GSFC).

Other guidelines to provide format uniformity and data correlation include the following:

- o Minor frame length should be the same for all instruments;
- o Except for single bit (on/off measurements) data from different measurements should not be multiplexed except in 8 bit bytes;
- o Symmetrical commutation;
- o GMT should occur at a fixed location for any minor frame format;

- Digitized word length should be multiples of 8 bits.
- AMPS data required by the Orbiter PCM system should have sample rates that are multiples or submultiples of Orbiter samples;
- Since both the Spacelab and Orbiter data system do not operate on the traditional first-in/first-out data, data requiring precise time correlation will require; a) time tagging at the source or b) the time delay between the time the data was accessed and the time it was inserted into the data stream must be known. This delta time can then be used by the ground for time correction;
- All data should be referenced to the Orbiter Master Timing Unit.

Table 5.6.2-1 Telemetry Format Requirements

Parameter	GSFC ADS X-560-63-2	Shuttle Telemetry Data Format Control Book (JSC)
Word Length	32 Bits or Less	
Minor Frame Length	8192 Bits or Less	Multiple of 16 Bits; 8192 Bits or Less
Major Frame Length	256 Minor Frames or Less	
Submultiplex	Submultiplexer Cycle to be Complete Within One Major Frame	
Format Identification	Positive Identification of Variable Formats	
Frame Synch	7 to 30 Bits Code Pattern/Frame	Synch at Beginning of Minor Frame. Bit Length to be 8, 16, 24 or 32 Bits
Format Identification	---	Required
Payload Identification	---	Required
Bit Rate Identification	---	Required
Format Change		To Occur at Beginning 1st Minor Frame Belonging to First Major Frame

5.6.2.2 Discussion

Analysis of AMPS data reveals diverse format requirements whose bit rates are from a few thousands to over 3.8 megabits. These data may originate from instruments mounted on the pallet, or from deployed packages. Figure 5.6.2-1 represents the data source, routing and subsequent interleaving of the required data. All data are grouped into 8 bit word lengths or multiples thereof. Data from a pallet instrument to the experiment I/O-computer is limited to 100 Kbps on AMPS and consists of low rate science data, instrument feedback data required by the computer or experiment operator or status data required by the Orbiter crew. A second output from the pallet instrument to the high rate multiplexer is provided for the sole purpose of acquisition and interleaving of the instrument science data plus the necessary corollary data required for post flight processing. This data rate will be from 80 Kbps to 4.75 Mbps. While data interfacing directly with the experiment I/O-computer may not need all of the overhead data shown, this is typical of data to be inserted into the data format which is routed into the high rate multiplexer.

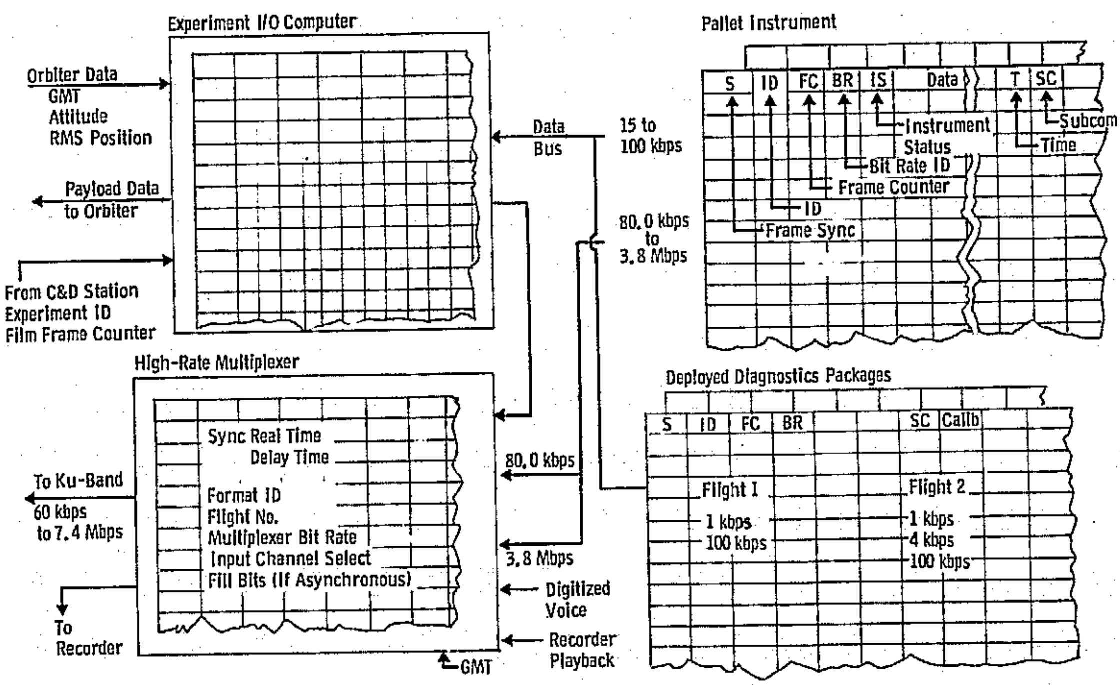


Figure 5.6.2-1 Telemetry Data Interleaving and Data Correlation

Data which are generated by the RMS mounted diagnostic packages or free-flying subsatellites will output standard telemetry formats. The data rates will vary from 1 Kbps to 100 Kbps. As an example, the characteristics of the data from the Electron Beam Diagnostic Package

have been developed and are shown in Table 5.6.2-2. This data is formatted into a minor frame length of 250 eight-bit words (2000 bits/minor frame). Table 5.6.2-2 shows two major frame lengths of 50 and 25 minor frames. Two options are available once the data is received by the Spacelab. One is to route the data into the RAU while the second is to route the data both to the RAU and the high rate multiplexer. The latter approach requires the computer to process only that data required by the onboard crew while the former approach requires an additional function of throughputting the data from the experiment I/O-computer to the high rate multiplexer. The examples shown use a minor frame length of 2000 bits although frame lengths of 8192 bits for the minor constituent experiments may be desirable.

Table 5.6.2-2 Electron Beam Diagnostic Telemetry Format

Instrument	Data Length (Bits)	Bits Per Second	12500 Word Format*	6250 Word Format**
			No. Words	No. Words
Fluxgate	32	3,200	396	198
Electrostatic Analyzer	18	80,000	9,892	4,945
Cold Probe	16	480	59	30
OBIPS	16	9,600	1,187	593
Housekeeping	8	5,000	618	309
Faraday Cup	16	1,600	198	100
Subtotal		99,880		
Sync	24		150	75
TOTAL			12,500	6,250

Word Length = 8 Bits

Minor Frame = 250 Words (2000 Bits)

*Major Frame = 50 Minor Frames
1 Major Frame/Second

**Major Frame = 25 Minor Frames
2 Major Frames/Second

The need for major frames are primarily a function of subcommutation requirements, i.e., if no subcommutation were required, there would be no need for major frames. For AMPS, subcommutation of housekeeping measurements will be required for efficient formatting and word slots should be allocated for this function. Supercommutation will also be required, especially for the minor constituent experiments whose data rates exceed 1 Mbps.

Data correlation will require word slots for GMT/MET, and the use of the 1024 KHz and 4 pps is available for greater precisions. GMT is the one reference point common to the Orbiter, Spacelab and AMPS instrument data as illustrated in Figure 5.6.2-2. The insertion of GMT into the instrument data, pointing computer data and the experiment I/O-computer assures the capability for precise data correlation. In addition, the figure illustrates the complexity of data transfer between the experiment I/O-computer and the pointing platform computer for platform operation and data acquisition. The Orbiter data (Table 5.6.2-3) required by the platform computer is also interleaved with the Spacelab/AMPS data to assure that prime data required for post flight analysis is on one data stream.

The formatting and data processing requirements of the experiment computer are also illustrated by Figure 5.6.2-1. As this data is routed to the high rate multiplexer, the multiplexer must insert its own header (overhead) which amounts to synch words on top of synch words. Not only is real time synch required but delay time synch is also required if data is recorded. The arrangement of synch and other overhead data versus word slot should be fixed for any bit rates. Other overhead data includes format I.D., input channel select, and bit rate I.D. which are required for efficient decommutation. Should the high rate multiplexer to experiment interface be asynchronous additional overhead data in the form of fill bits will be required. In an asynchronous system the sampling rate of the experiment data is slower than the high rate multiplexer clock rate. In such a system, fill words will be required along with bit identification techniques provided to distinguish between valid data and fill data. Fill words are necessary to preserve format symmetry since a valid data word will not always be ready at the multiplexer sampling time.

5.6.2.3 Conclusion

Guidelines for telemetry format are provided and must be established early in the AMPS program to provide a low cost approach to post flight data analysis. These guidelines are consistent with NASA documentation on telemetry formats. In particular every instrument format must provide proper header data to facilitate ground data cataloging and processing. Measurements requiring precise time correlation must have time skew correction factors inserted in the post flight data analysis. In addition, all data must be referenced to the Orbiter timing system. Interleaving as much of the required Orbiter data into the AMPS/Spacelab telemetry data stream will provide for efficient and independent data reduction.

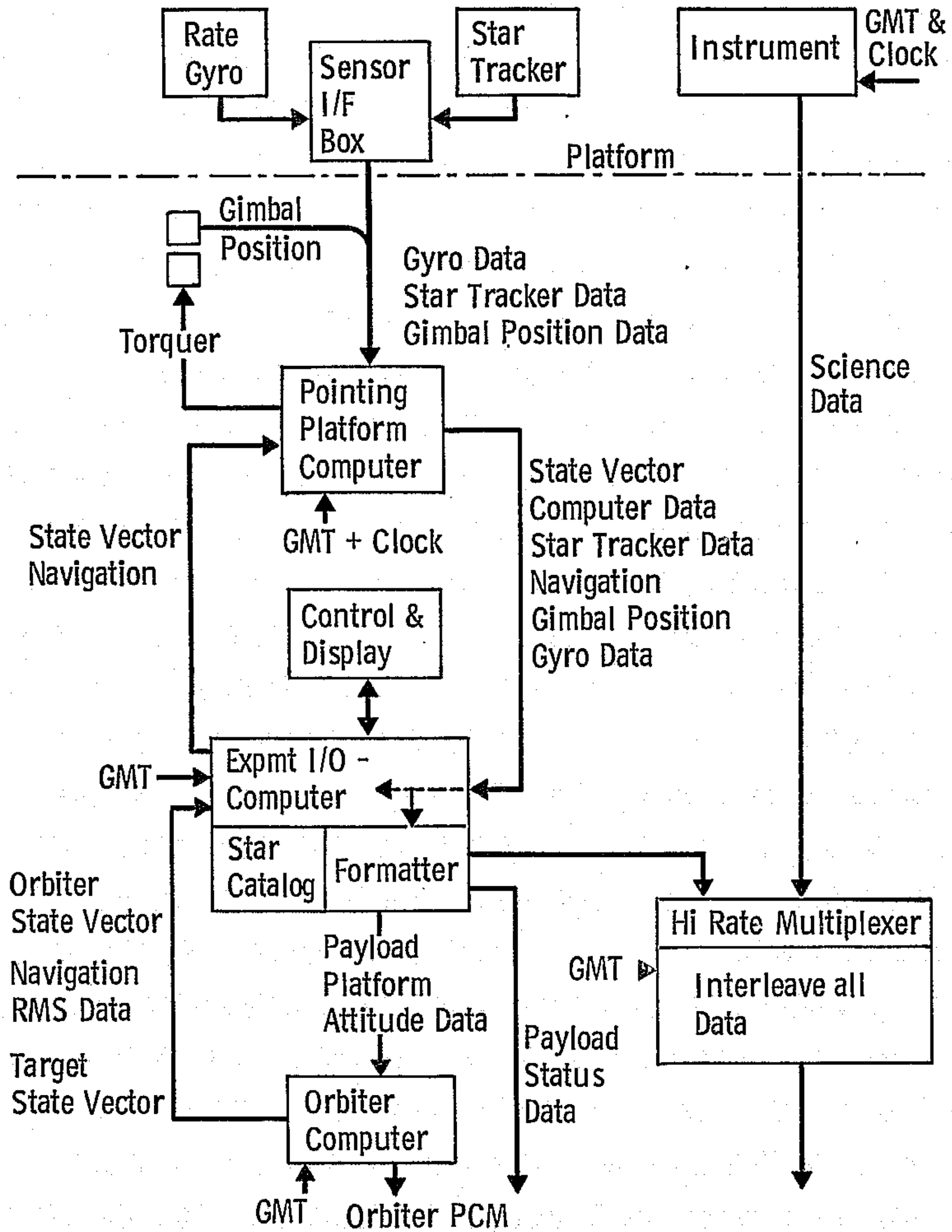


Figure 5.6.2-2 Pointing Platform Data Traffic and Correlation

Table 5.6.2-3 Orbiter Computer to AMPS Data Transfer

COMMAND 1 ORBITER STATE VECTOR UPDATE - INERTIAL VERTICAL

	Words
16 Bit Computer Data Words Containing:	
Orbiter Position Vector* - 3 Axis, 48 Bits/Axis	9
Velocity Vector - 3 Axis*, 48 Bits/Axis	9
MET - Relative To T-O, 48 Bits	3
GMT - Relative to Greenwich Meridian, 48 Bits	3
Attitude Information - 3 Gimbal Angles (16 Bits Each)	3

*With Respect to Earth Centered Inertial (ECI) Coordinate System

COMMAND 2 ORBITER STATE VECTOR UPDATE--LOCAL VERTICAL

Same as Command 1 Except Orbiter Attitude Data is Relative to Local Vertical Rather Than Inertial Vertical.

Attitude Information - 3 Euler Angles (Azimuth, Elevation, Roll)	3

COMMAND 3 TARGET STATE VECTOR UPDATE

Target Can Be Rendezvous Point, An Orbiting Object, a Future State Vector Application, Etc.

16 Bit Computer Data Words Containing:	Words
Target Position Vector - 3 Axis*	9
Target Velocity Vector - 3 Axis*	9
GMT	3

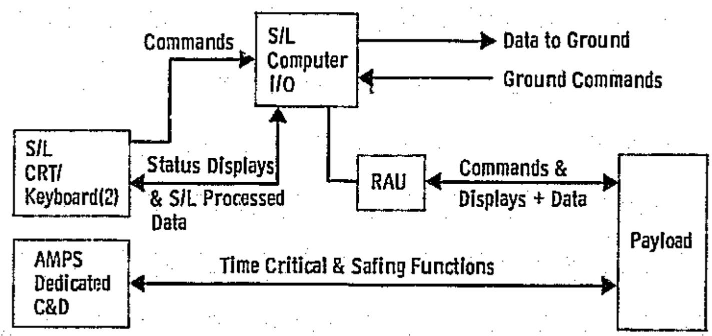
*In ECI Coordinates

5.7 Control & Display Subsystem

5.7.1 Dedicated vs Computer Interactive Controls and Displays Trade Study

This trade study was performed early during the Phase B study to evaluate the advantages of using Spacelab provided hardware and corresponding overall C&D implementation (computer interactive), as opposed to more conventional payload dedicated hardware and to select the approach to be used during the preliminary design phase.

Two C&D approaches were configured, differing primarily in the extent of their interfaces with the Spacelab CDMS and the quantity of AMPS unique C&D hardware. The alternate approaches and their key features are shown in Figures 5.7.1-1 and -2.



Key Features

1. Configured for Maximum CDMS Interface
2. Minimum new hardware - small dedicated C&D, cabling, etc
3. Command/display processing performed by S/L computer; modified via CRT/KB or ground
4. Maximum software required
5. Modular software applications packages; standard RAU interface; all functions available at RAU/payload interface

Figure 5.7.1-1 Computer Interactive C&D Concept

Table 5.7.1-1 presents an evaluation of 16 parameters grouped into three areas: (1) crew; (2) operations; and (3) hardware. Although cost was not broken out as a distinct parameter, each item was weighted and ranked with cost as a driving variable. Rank (1 or 2), for each concept, was determined by assessing the relative merits of the concept in relation to the parameter. The more desirable concept was given the higher rank. The weighting factor (1-4) for each parameter reflects the relative importance of the parameter with respect to the AMPS program objectives of maximum flexibility with minimum cost. The parameters with greater

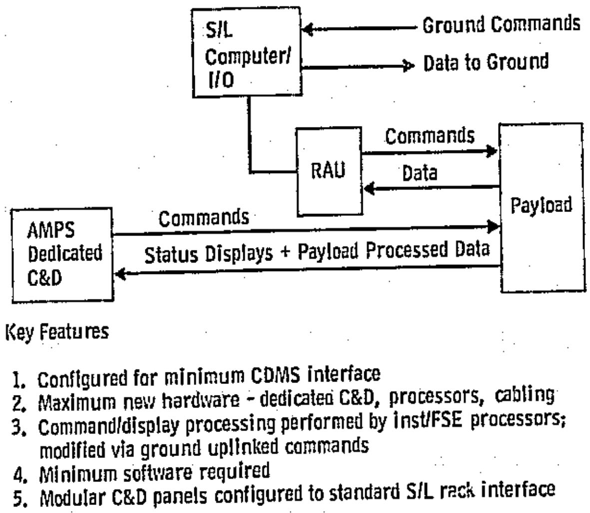


Figure 5.7.1-2 AMPS Dedicated C&D Concept

impact on the objectives were given higher weights. A brief rationale is also presented for each parameter.

The results of the trade study clearly indicate the advantages of the computer interactive concept (83) over the dedicated C&D concept (49) in all three categories. Therefore, the computer interactive concept, which uses maximum Spacelab capability, was used for preliminary design activities.

Table 5.7.1-1. Concept Trade Evaluation

Trade Parameter	Weight (1-4)	Computer Interactive Concept Rationale	Rank/Score	AMPS Dedicated C&D Concept Rationale	Rank/Score
Crew					
Workload	2	High Flexibility; Crew/Ground Functional Allocation Can Be Modified Real Time	2/4	Inflexible; Crew/Ground Functional Allocation Established Premission	1/2
Training	2	Maximum Training for CRT/Keyboard Operations	1/4	Minimum Training Due to Conventional Man/Machine Interface	2/4
Operator Error Probibility	3	Low - All Commands Can Be Displayed & Verified Before Entry	2/6	High - Possible Inadvertent Control Activation & Lack of Command Verification Before Entry	1/3
Information Presentation	3	Very Flexible; Recognition & Response Times Can Be Optimized	2/6	Limited to Available C&D Hardware Components (Switches, Flags, Meters, etc)	1/3
Task Performance Time	4	Centralized Data Entry with Reduced Links; Format Flexibility Reduces Interpretation Requirements & Aids in Decision Making	2/8	Increased Eye/Hand Links & Component Operation Times	1/4
Crew Station Geometry	4	Compact; Optimum Use of Human Engineering Design Criteria for Control Panel Layout & Crew Station Design	2/8	Potential Panel Layout Difficulties Due to Area & Volumeetric Requirements; Detailed Analyses Required for Each Flight	1/4
Operations					
In-Flight Automatic Sequence Modification	4	Yes - Crew & Ground	2/8	Yes - Ground Only	1/4
Ground Uplink Command	3	Yes - Flexible	2/6	Yes - Fixed	1/3
Equipment Characteristics					
Power, Weight Volume	1	Low - Minimum Additional Hardware	2/2	High - Dedicated C&D Components Required	1/1
Complexity	3	Maximum Complexity - Sophisticated Hardware/Software Interfaces between CRT/Keyboard/Data Bus/RAU	1/3	Minimum Complexity - Straight-Forward Proven Approach	2/6
Physical Interfaces	2	Minimum - Data Bus for Majority of Command/Display Interfaces	2/4	Maximum Physical Interfaces between C&D Panels & S/L plus Maximum Interface Cabling	1/2
Standardization	3	Standardized Software Application Modules; Development & Integration Software Aids Available	2/6	Standardize Panel/Rack Interface; Panels & Interfaces will Change across Missions	1/3
Flexibility	4	Majority of Changes via Software	2/8	Majority of Changes Require Redesign & Hardware Modification & Build	1/4
Growth	3	Limited by Core/Mass Memory Capability	2/6	Limited by S/L Module Mounting Capabilities & Feedthrough Constraints	1/3
Instrument/FSE	3	Minimum - Straightforward Interface; All Functions Available at Payload/RAU Interface	2/6	Requires Dedicated Payload Processors	1/3
		Total Score	83		49

5.7.2 Task Analyses

Preliminary task analyses were performed for the following AMPS experiments: (1) Acoustic Gravity Waves; (2) Electron Beam Studies; and (3) Laser Sounder. The results of these analyses provided inputs to the concept trade study (Section 5.7.1) and provided a sample operating procedure for an experimenter/computer dialogue simulation (Section 5.7.3) for the Laser Sounder.

Task analyses are necessary to define the experimenter/payload interface and verify the experiment C&D functional requirements. The analyses identify information requirements, feedback loops, experimenter limitations, missing and redundant functions, and incompatible or omitted information links. Preliminary task analyses provide qualitative information with respect to experimenter work load. Detailed task analyses, supported by high fidelity simulations, provide quantitative information with respect to actual task performance times, mandatory for preparing experiment and mission timelines.

Detailed functional designs were developed for each of the experiments identified previously. Figures 5.7.2-1 and 5.7.2-2 define the Laser Sounder experiment configuration as an example of the analyses performed. The principal command and display requirements, shown in the figures as inputs and outputs, were expanded to identify detailed C&D functional requirements. The functional requirements, together with hardware and operational constraints, were then used to define the experimenter tasks. For each task, the control and display hardware and the experimenter action and information feedback requirements were identified.

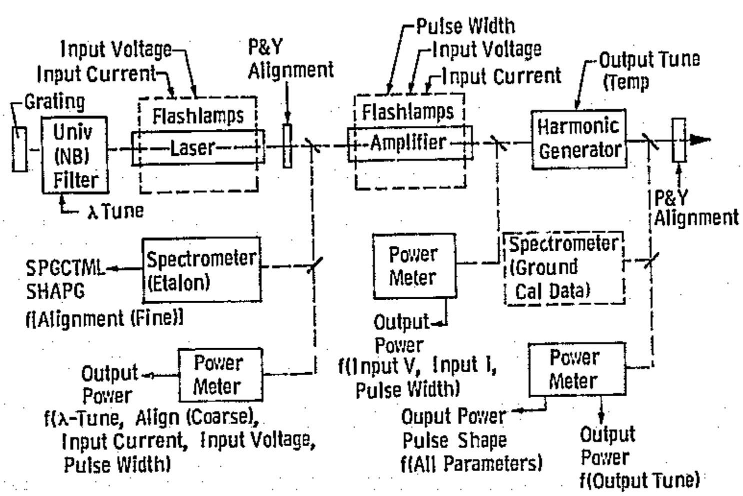


Figure 5.7.2-1 Laser Transmitter - Narrow Band

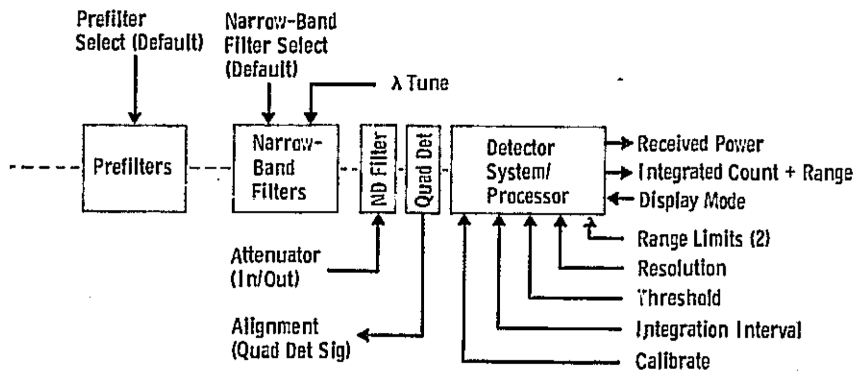


Figure 5.7.2-2 Laser Receiver

Table 5.7.2-1 shows the initial task analysis for the Laser Sounder experiment. We assumed manual operation for each task in order to verify the functional C&D requirements with respect to the task analysis parameters. This task analysis was then used as a sample operating procedure for an experimenter/computer dialogue simulation. The simulation resulted in the automation of a number of task sequences in order to minimize the experimenter work load and assign, to the computer, those tasks which it could perform more accurately and efficiently. Table 5.7.2-2 shows the modified task analysis. Similar analyses were performed for the two remaining experiments to aid in the selection of the C&D approach for AMPS and to support software sizing.

Table 5.7.2-1 LIDAR Task Analysis

Function	Control	Display	Action	Feedback
Select LIDAR Experiment	FKB A/N KB	CRT CRT	Enter "Exp" Enter LIDAR (OH) Line Number	Lists Available Experiments LIDAR Setup Format
Turn on Trans Main Power	A/N KB	CRT	Enter Line Number + On	Main Power Status = On
Open Trans Apertur Door	A/N KB	CRT	Enter Line Number + Open	Aperture Door Status = Open after TBD-Second Delay
Turn on Trans Dye Pumps	A/N KB	CRT	Enter Line Number + On	Dye Pumps Status = On
Turn on Trans Thermal Control	A/N KB	CRT	Enter Line Number + On	Thermal Control Status = Ready after TBD-Second Delay
Turn on Receiver Main Power	A/N KB	CRT	Enter Line Number + On	Main Power Status = On
Open Receiver Aperture Door	A/N KB	CRT	Enter Line Number + Open	Aperture Door Status = Open after TBD-Second Delay
Turn on Receiver Thermal Control	A/N KB	CRT	Enter Line Number + On	Thermal Control Status = Open after TBD-Second Delay
Turn on Receiver High Voltage	A/N KB	CRT	Enter Line Number + On	High-Voltage Status = On
Verify Setup Sequence Completed	—	CRT	Observe Experiment Status Indicator	Status = Ready
Access Calibration Sequence	FKB	CRT	Enter "Cal"	Calibration Format
Adjust Laser Internal Alignment	A/N KB FKB	CRT CRT	Enter Line Number + On Adjust Alignment P&Y for Maximum Power	Alignment Function = On Power Output Level
Adjust Laser Wavelength	A/N KB FKB	CRT CRT	Enter Line Number + On Adjust Wavelength for Maximum Power	Wavelength Function = On Power Output Level
Set Laser Input Voltage Level	A/N KB	CRT	Enter Line Number + Value	Input Voltage Level
Set Laser Input Current Level	A/N KB	CRT	Enter Line Number + Value	Input Current Level
Adjust Laser Output Spectral Shape	A/N KB FKB	CRT CRT	Enter Line Number + On Adjust Alignment	Alignment Function = On Spectral Shape - Plot
Set Amplifier Input Voltage Level	A/N KB	CRT	Enter Line Number + Value	Input Voltage Level/Gain
Set Amplifier Input Current Level	A/N KB	CRT	Enter Line Number + Value	Input Current Level/Gain
Adjust Harmonic Generator Pulse Shape	FKB	CRT	Adjust Harmonic Gen Temp Adjust Pulse Width	Pulse Shape - Plot
Align Transmitter to Receiver	A/N KB FKB	CRT CRT	Enter Line Number + On Adjust Alignment P&Y	Alignment Function = On Alignment Display Status
Calibrate Receiver Detector	A/N KB FKB	CRT CRT	Enter Line Number + On Enter "Start"	Detector Cal Function = On Status = "Ready" after TBD-Second Delay
Tune Receiver	A/N KB FKB	CRT CRT	Enter Line Number + On Adjust Wavelength for Maximum Power	Receiver Power Function = On Receiver Power Level
Access LIDAR Operating Sequence	FKB	CRT	Enter "Data"	Operating Format
Verify Experiment Status	—	CRT	Observe Status Flag	Status = "Ready"
Verify/Adjust Operating Parameters	A/N KB	CRT	Enter Line Number + New Parameters Value	Modified Parameter Values
Initiate Operation	FKB	CRT	Enter "Start"	Status = "Operate"
Analyze Data	—	CRT	Observe Data Display	Plot - Counts vs Altitude
Readjust Operating Parameters As Required	FKB A/N KB	CRT CRT	Enter "Stop" Enter Line Number + New Parameter Value	Status = "Ready" Modified Parameter Values
Iterate Operation, Analysis, and Parameter Adjustment Until Data Is Satisfactory				
Terminate Operation	FKB	CRT	Enter "Stop"	Status = "Ready"
Experiment Can Be Initiated by a Computer Driven Command According To a Preprogrammed Flight Plan				

Table 5.7.2-2 LIDAR Task Analysis Rev. A

Function	Control	Display	Action	Feedback
Select LIDAR Experiment	FKB A/N KB	CRT CRT	Enter "Exp" Enter LIDAR Line Number	List of Available Expts LIDAR Setup Format
Initiate Auto Setup Seq	FKB	CRT	Enter "Start"	Flashing Status Indicator Advise In-Process Functions
After ~15 sec Sequence is Complete	--	CRT	Observe Setup Status	Setup Status = "Ready"
Access Calibration Sequence	FKB	CRT	Enter "Continue"	Calibration Status Format
Initiate Automatic Calibration Sequence	FKB	CRT	Enter "Start"	Flashing Status Indicators Advise In-Process Functions
Align Laser				
Adjust Laser Wavelength			Automatic	
Adjust Laser Input Voltage/Current				
Adjust Laser Output Spectrum	FKB	CRT	Fine Tune Alignment then "Continue"	Spectral Shape - Plot
Adjust AMP Input Voltage/Current			Automatic	
Set Harmonic Gen Output Power	A/N KB	CRT	Input Desired Power	Power Level - Numeric
Adjust Output Pulse Shape	FKB	CRT	Adjust Pulse Width & Temp	Pulse Shape - Plot
Align Transmitter to Receiver				
Calibrate Receiver Detector			Automatic	
Tune Receiver Power				
Access Operating Sequence	FKB	CRT	Enter "Data"	Operating Format
Verify Experiment Status	--	CRT	Observe Status Flag	Status = "Ready"
Verify/Adjust Operating Parameters	A/N KB	CRT	Enter Line Number + New Parameter Value	Modified Parameter Values
Initiate Operation	FKB	CRT	Enter "Start"	Status = "Operate"
Analyse Data	--	CRT	Observe Data Display	Plot - Counts vs Altitude
Readjust Operating Parameters as Required	FKB A/N KB	CRT CRT	Enter "Stop" Enter Line Number + New Parameter Value	Status = "Ready" Modified Parameters Values
Iterate Operation, Analysis, & Parameter Adjustment until Data satisfactory				
Terminate Operation	FKB	CRT	Enter "Stop"	Status = "Ready"
Experiment Can Be Initiated by Computer-Driven Command According to Preprogrammed Flight Plan				

5.7.3 Experimenter/Computer Dialogue Analysis

The objectives of this analysis were to:

- o Evaluate various types of experimenter/computer dialogues with respect to AMPS operational requirements
- o Investigate the implementation of the Spacelab function keyboard
- o Identify the capabilities, limitations, and constraints imposed on AMPS payload operations by the Spacelab CDMS hardware configuration

A computer simulation of a typical AMPS experiment operating sequence was developed to provide an engineering tool for accomplishing these objectives. The simulation hardware is shown in Figure 5.7.3-1. The Spacelab CRT, simulated on the left portion of the screen, can display alphanumeric data (21 lines, 47 characters per line), vectors, and graphics. The tri-color capability of the Spacelab CRT was not simulated. The right portion of the CRT display simulates the Spacelab 25 key function keyboard. For simulation purposes only, the function keyboard is a light pen sensitive screen area instead of separate hardware.

Various dialogues were investigated for use by AMPS payloads, but only those dialogues compatible with AMPS and Spacelab hardware were considered. This constraint eliminated such alternatives as light pen inputs and finger touch displays.

Dialogues fall into two basic categories: (1) user initiated, and (2) computer initiated. Interaction speed and accuracy are principal requirements of Spacelab payload dialogues due to the short mission duration and generally complex payload operational requirements. User initiated dialogues, where the computer responds to the user, usually require that the user be highly trained and proficient in computer system operations and also familiar with the details of payload operations. We do not feel that this type of training should be required of the AMPS experimenter since his background is science oriented. We feel that a computer initiated dialogue would alleviate the experimenter from the responsibility of operational details and allow him to concentrate on the scientific aspects of payload operations. Operational details would be relegated to the computer system and supported by ground personnel. However, for a well designed computer system with modular user interface software, this does not preclude the incorporation of an additional, optional, user initiated dialogue which can be made effective for the well-trained user and also can be invisible to other users.

Three of the simplest and most effective computer initiated dialogues are: (1) menu selection; (2) form filling, and (3) simple operator instructions. The menu selection approach applies when only a limited set of valid answers exist to a computer initiated question. In

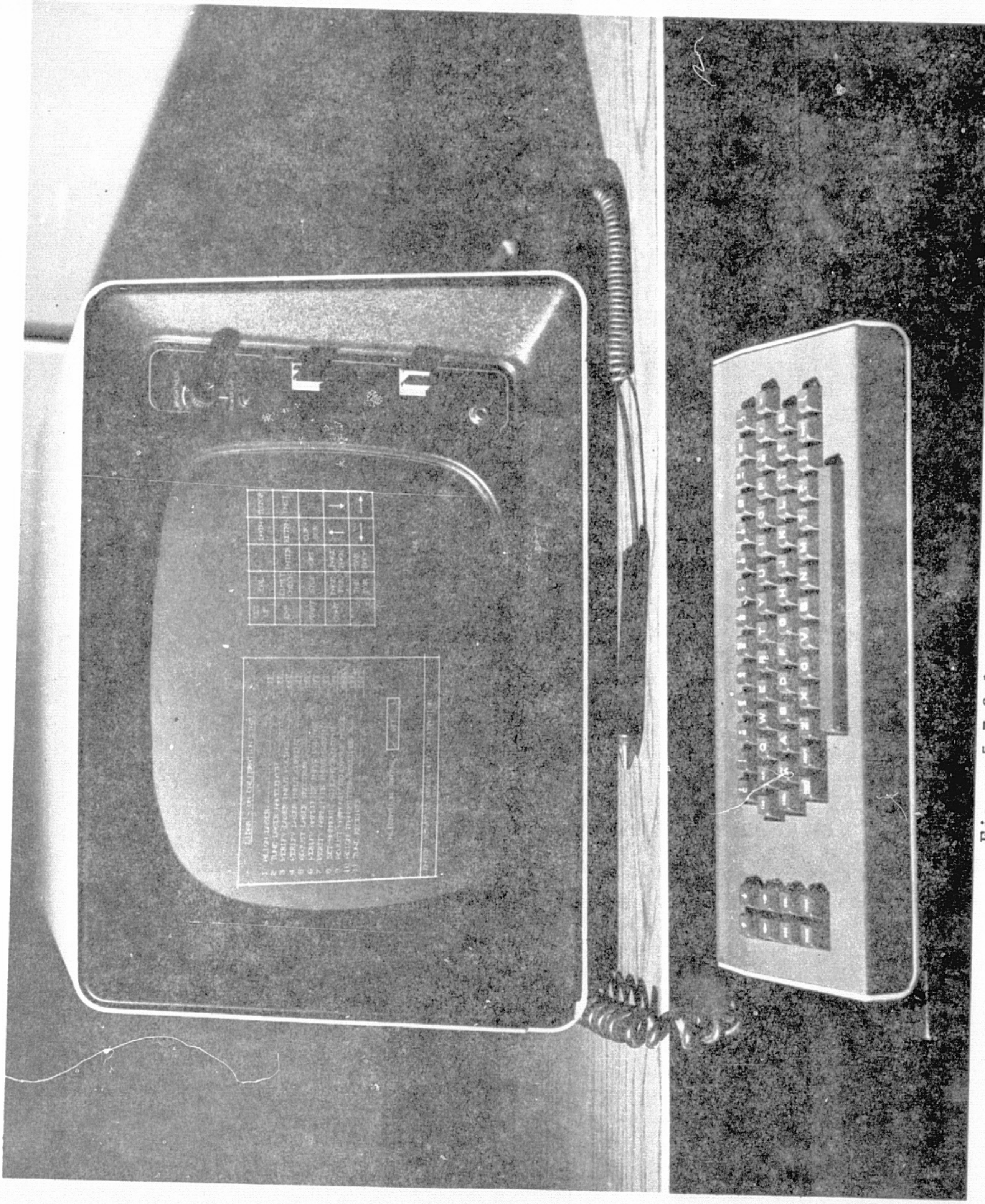


Figure 5.7.3-1 Simulation Hardware

that case, the user selects one answer from a displayed list. Figure 5.7.3-2 is an example of a menu selection format. The choice is indicated by entering the line number of the desired experiment using the alphanumeric keyboard. This is the fastest method for a menu selection input. Other methods such as tabbing the cursor to the desired entry and entering an "X" in a box, although less error prone, are more time consuming.

Experiment Status	
1. Acoustic Gravity Waves	Off
2. Electron Beam Studies	Off
3. Minor Constituents	On
4. EMI Field Mapping	Off
5. Particles/Gas Effluents	Oper
6. Wake Mapping	Off
Select Experiment Number:	

Figure 5.7.3-2 Menu Selection Format Example

The form filling technique for data entry presents the user with a form with blanks to be filled in to identify the desired data or variable value. The entry is made by entering the line number of the variable and then the desired value using the alphanumeric keyboard.

An improvement on the form-filling technique is the use of default parameter values. This consists of selecting premission baseline parameter values and allowing the experimenter to accept or modify the default variable values based on actual experiment performance. An example of a typical display format of this type is shown in Figure 5.7.3-3. Lines numbered 1 to 5 identify the variables and their default values (underlined). The experimenter can modify the values by entering the line number and the new value.

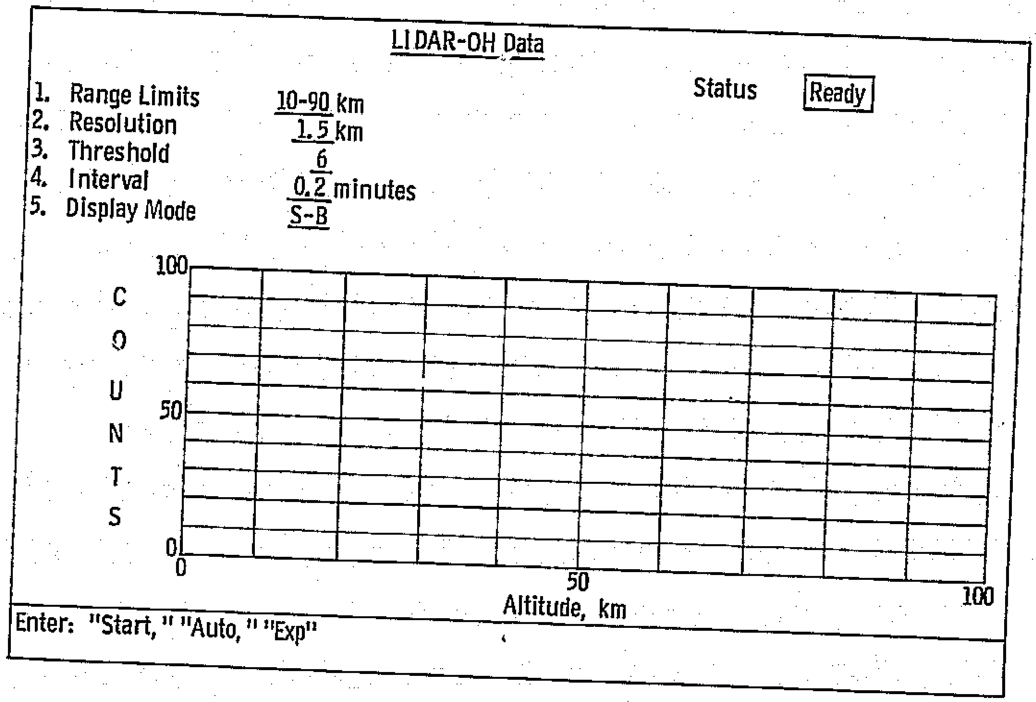


Figure 5.7.3-3 Form Filling/Simple Users Instruction Format Example

In the "simple user instruction" technique the computer displays an instruction to the user together with the acceptable responses. An example of a display format for this technique is shown in Figure 5.7.3-3. The bottom lines list the principal responses; and the choice is indicated by typing in the desired response using the alphanumeric keyboard.

The function keyboard is a useful supplement to the alphanumeric keyboard, particularly for inputting "simple user instruction" commands which are frequently used. Figure 5.7.3-4 shows the AMPS simulation keyboard layout. The first two lines of keys are used to provide quick access to experiment and system status and data from any level of operation. Keys are provided for quick access to experiment status (EXPER), setup (SETUP), calibration (CAL) and data (DATA), instrument and subsystem (INST/SS) status and default parameters, special operational procedures (TUTOR), variable options (OPTIONS), and the automatic limit check program variables (LIMIT CHECK). Keys are also provided to initiate (START) and terminate (STOP) automatic sequences, to branch from (INTERRUPT) and return to (RETURN) both manual and automatic sequences, to step through preprogrammed task sequences (CONT), control cursor position (↑, ↓, ←, →, CURSOR HOME), and under specific conditions which depend on the task being performed, to input analog commands (↑, ↓, ←, →).

EXPER	INST	SETUP	CAL	DATA
TUTOR	OPTIONS	LIMIT CHECK		
START	STOP	PAGE FORWARD	↑	PAGE BACK
INTERRUPT	RETURN	←	CURSOR HOME	→
CONTINUE		TAB FORWARD	↓	TAB BACK

Figure 5.7.3-4 Function Keyboard Layout

The functions assigned to the simulation function keyboard were selected for the specific purpose of evaluating the potential keyboard usage. A detailed analysis of several payloads will be required before the functions can be fully optimized.

The results of the dialogue analysis and simulation activities indicate that the dealogue finally used for AMPS will be a mixture of several dialogues in order to optimize the man/computer interface with respect to communication speed and accuracy and to optimize the use of available Spacelab capability. The analysis and simulation results at this time indicate the following:

(1) Alphanumeric keyboard inputs are undesirable since they are time-consuming and have a high error probability.

(2) The menu selection approach is a simple but effective method for inputting commands quickly and with a minimum of errors.

(3) The Spacelab function keyboard is effective for inputting repetitive types of frequently used commands and is a desirable supplement to the alphanumeric keyboard.

(4) The Spacelab CRT has a restricted information presentation capability due to its physical size and will require highly optimized display formats in order to meet complex payload operational requirements.

5.8 Communication Subsystem

5.8.1 K_u Antenna Coverage Analysis

This analysis was performed to determine the extent of the K_u band antenna blockage for several Orbiter attitudes pertinent to AMPS and other Labcraft payloads. The COCOA (Computer Oriented Communications Operational Analysis) program was used to generate mission profiles of antenna look angles (theta and phi) between the Orbiter and the TDRSS for specific fixed orbiter attitudes. These data were then compared with K_u band antenna blockage envelopes to determine periods during which antenna blockage existed.

The Orbiter provides a K_u band RF system for the communication of wideband data via the TDRSS. Included in this system is a steerable, 20-inch diameter, parabolic antenna which is mounted at the end of a short boom and deployed from the cargo bay. The baseline system provides one boom-antenna combination, but an option is available to deploy another boom and antenna from the opposite side of the bay. The baseline, or primary antenna, is deployed to the right side of the Orbiter looking from the cargo bay toward the cockpit; and the kit antenna is deployed to the left side. Front and top views of the deployed primary antenna are shown in Figures 5.8.1-1 and -2, which also indicate the primary antenna coordinates. The kit antenna X and Z coordinates are the same, with the Y coordinate changing to the opposite polarity.

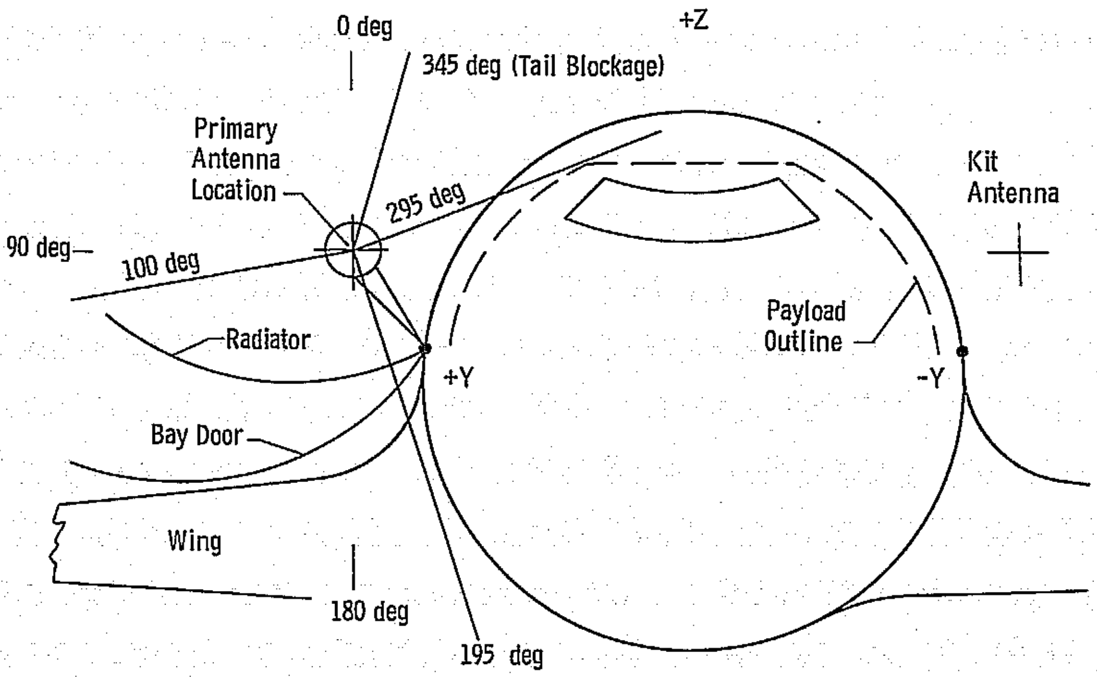


Figure 5.8.1-1 Antenna Blockage - Front View

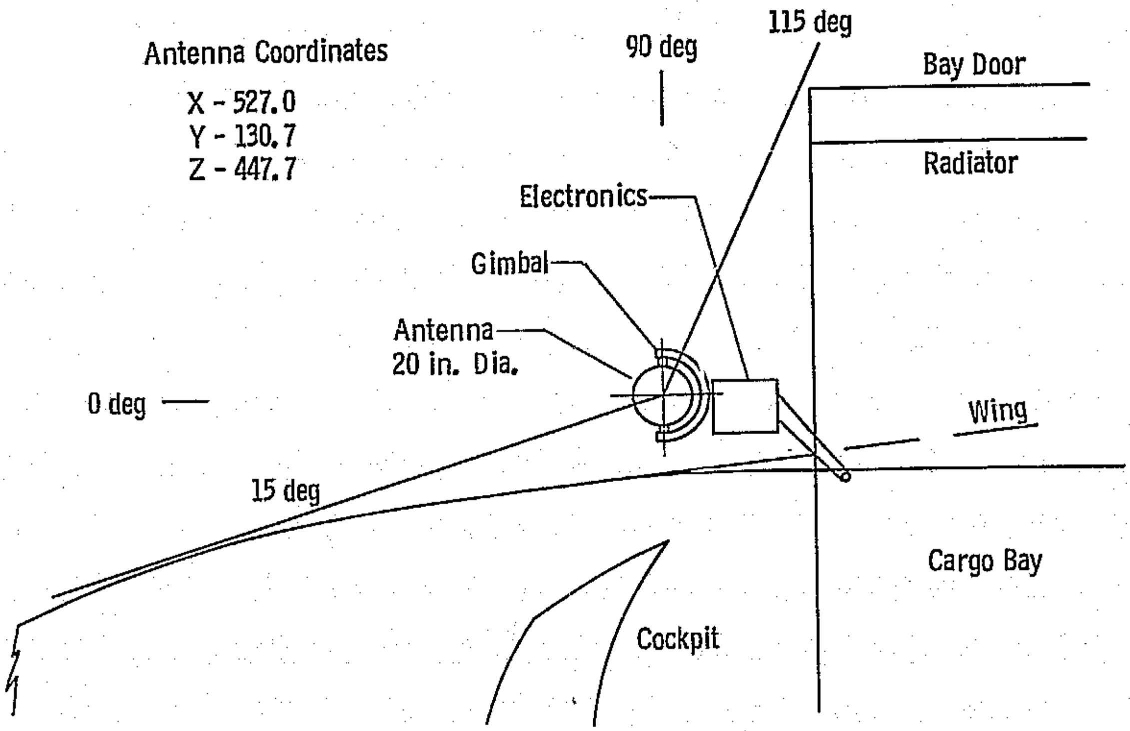


Figure 5.8.1-2 Antenna Blockage - Top View

It becomes obvious from these figures that the antenna is deployed very close to the Orbiter body, and, consequently, suffers blockage which reduces its effectiveness. Depending on Orbiter attitude, this blockage can be quite severe, and can reduce the usefulness of this system to less than half the mission time. These figures also help to define the antenna blockage envelope, which is that combination of theta and phi look angles through which antenna line-of-sight to a TDRS is blocked. Basically, these envelope limits are independent of vehicle attitude since these measurements are made with respect to the vehicle coordinate system. However, to avoid generating different computer data for attitudes where only a rotation about the vehicle X axis is involved, such as rotating from a Z-LV attitude to either a +Y or -Y-LV, it is possible to rotate the envelope phi angle limits rather than generate computer data for the new attitudes. This procedure was used for analyzing the antenna blockage associated with three possible AMPS attitudes; and the envelope limits are listed in Tables 5.8.1-1 and -2 for the two Orbiter antennas.

Table 5.8.1-1 Primary Antenna[1] Blockage Angles

θ Limits	Ø Limits			Blockage Elements
	Z-LV	-Y-LV	+Y-LV	
$115^\circ < \theta < 170^\circ$	$90^\circ < \phi < 295^\circ$	$180^\circ < \phi < 25^\circ$	$0^\circ < \phi < 205^\circ$	Bay Doors and Radiators
$40^\circ < \theta < 115^\circ$	$195^\circ < \phi < 295^\circ$	$285^\circ < \phi < 25^\circ$	$105^\circ < \phi < 205^\circ$	Orbiter Forward Section
$15^\circ < \theta < 40^\circ$	$195^\circ < \phi < 270^\circ$	$285^\circ < \phi < 360^\circ$	$105^\circ < \phi < 180^\circ$	
$\theta > 170^\circ$	$195^\circ < \phi < 345^\circ$	$285^\circ < \phi < 75^\circ$	$105^\circ < \phi < 255^\circ$	Tail Section
$\theta < 15^\circ$	No Blockage	No Blockage	No Blockage	----

1 - Located to the right of the Orbiter looking from the payload bay toward the Orbiter cockpit

Table 5.8.1-2 Kit Antenna[1] Block Angles

θ Limits	Ø Limits			Blockage Element
	Z-LV	-Y-LV	+Y-LV	
$115^\circ < \theta < 170^\circ$	$65^\circ < \phi < 270^\circ$	$155^\circ < \phi < 360^\circ$	$335^\circ < \phi < 180^\circ$	Bay Doors and Radiators
$40^\circ < \theta < 115^\circ$	$65^\circ < \phi < 165^\circ$	$155^\circ < \phi < 255^\circ$	$335^\circ < \phi < 75^\circ$	Orbiter Forward Section
$15^\circ < \theta < 40^\circ$	$90^\circ < \phi < 165^\circ$	$180^\circ < \phi < 255^\circ$	$0^\circ < \phi < 75^\circ$	
$\theta > 170^\circ$	$15^\circ < \phi < 165^\circ$	$105^\circ < \phi < 255^\circ$	$285^\circ < \phi < 75^\circ$	Tail Section
$\theta < 15^\circ$	No Blockage	No Blockage	No Blockage	----

[1]Located to the left of the Orbiter looking from the payload bay toward the Orbiter cockpit

As this analysis and the Phase B study progressed, it became apparent that some Labcraft missions would be flown primarily in a solar or stellar inertial attitude, or in some variation of these attitudes. Therefore, data were also generated for specific solar and stellar attitudes, but not considering any variation such as a fixed rotation about one Orbiter axis off the primary attitude. The same basic envelope data as shown for the AMPS Z-LV attitude was applied to this solar and stellar inertial attitude data.

The existing COCOA program was used to generate look angle profiles for typical AMPS and Labcraft missions. The AMPS data were generated for a circular trajectory having an altitude of 250 KM and a 57 degree inclination; and for the +Z-LV, +Y-LV, and -Y-LV attitudes. Solar and stellar inertial attitude data were generated for circular trajectories at a 370 KM altitude and a 33 degree inclination. A solar inertial (+Z-SI) attitude was simulated; and two simulations with the Orbiter +Z axis held in stellar inertial attitudes, one anti-solar inertial (-Z-SI) and another with the +Z axis pointed perpendicular to the sunline (+Y-SI). A sample of the data is shown in Figure 5.8.1-3.

SATELLITE IDENT	SATELLITE LONGITUDE DEG	TIME OF CONTACT DAY HR MIN	SPACECRAFT LATTITUDE DEG	SPACECRAFT LONGITUDE DEG	SLANT RANGE N MI	SPACECRAFT LOOKANGLES THETA DEG	SPACECRAFT LOOKANGLES PHI DEG
		4 16 30.0	7.7	237.7	22551.3	121.1	33.0
		4 16 35.0	17.6	253.8	21613.2	119.7	34.4
		4 16 40.0	25.9	271.7	20789.2	119.3	36.1
		4 16 45.0	31.4	291.9	20178.9	119.9	37.9
		4 16 50.0	33.0	313.8	19864.9	121.3	39.5
	REVOLUTION NUMBER	70					
		4 16 55.0	30.3	335.3	19893.4	123.3	40.6
		4 16 60.0	23.9	355.0	20261.3	125.3	40.7
		4 17 5.0	15.1	12.3	20917.4	127.0	39.8
		4 17 10.0	4.9	28.0	21775.7	126.0	37.9
		4 17 15.0	-5.8	43.2	22735.0	128.1	35.3
		4 17 20.0	-15.9	59.1	23694.7	127.2	32.4
		4 17 21.3	-18.4	63.4	23932.2	126.8	31.6
TDRSH	189.000	4 17 27.9	-28.6	87.9	23700.5	116.4	185.8
		4 17 30.0	-30.7	96.4	23237.8	117.1	185.9
					SPACECRAFT ENTERED DRKNESS		
		4 17 35.0	-33.0	118.1	22088.7	118.3	185.9
		4 17 40.0	-31.1	139.9	20983.6	118.5	185.8
		4 17 45.0	-25.3	159.9	20054.3	117.6	185.4
		4 17 50.0	-16.8	177.6	19429.6	115.8	184.8
		4 17 55.0	-6.7	193.5	19207.9	113.5	183.8
		4 17 60.0	3.9	208.8	19427.9	111.2	182.7
		4 18 5.0	14.2	224.4	20054.4	109.5	181.4
					SPACECRAFT ENTERED DAYLT		

Figure 5.8.1-3 Typical Look Angle Data

Analysis Results - Results of this TDRSS coverage analysis are presented in Tables 5.8.1-3 and -4 in terms of a) cumulative coverage percentage; b) maximum continuous period of contact, and c) maximum continuous out-of-contact period, not considering brief periods of contact of less than 10 minutes. The above data include the regular periods of dropout between the two TDRSs attributable to the mission trajectory.

Table 5.8.1-3 KU Band - TDRSS Antenna Coverage

	Primary Antenna			Dual Antennas		
Parameter	Z-LV	+Y-LV	-Y-LV	Z-LV	+Y-LV	-Y-LV
Cumulative Coverage (per cent)	45	50	76	67	76.5	78
Maximum Contact (minutes)	38	27	85-90	85-90	85-90	85-90
Maximum Gap (minutes)	108	49	43	82	43	43

Z-LV Coverage Variations (Daily)

Antenna(s) \ Period	9 Rev (15 Hr)	5 Rev (8 Hr)	Cumulative
Primary	35%	60%	45%
Dual	57%	87%	67%

Table 5.8.1-4 Ku-Band Antenna Coverage, Solar/Stellar

Parameter \ Attitude	Solar (+Z - SI)			Stellar (-Z - SI)			Stellar (+Y - SI)		
	Primary	Kit	Dual	Primary	Kit	Dual	Primary	Kit	Dual
Cumulative Coverage, %	57	77	78	68	72	78.5	52	52	69.5
Maximum Contact Period, minutes	←	89	→	←	89	→	←	89	→
Maximum Gap, minutes	215*	44	44	44	91	44	280*	280*	45

*Marginal coverage for periods of 20 to 30 minutes.

As might be expected, a close evaluation of the blockage data reveals that there are cyclic periods where the coverage is relatively better or worse than the cumulative coverage figure. The Z-LV attitude, for example, can yield close to 60% coverage over a continuous 8 to 9-hour period; and can have its coverage deteriorate to about 35% over a 15-hour period considering only the primary antenna. The dual antenna configuration yields a range between 55 and 85%. A similar trend is noted for the +Y-LV, but the variations are not near as drastic. These trends are noted here because this can be a significant input to mission planning and analysis. It can also be concluded on the basis of the available data that the coverage profiles are approximately repetitive on a daily basis.

The analysis was concluded with a brief review of the above data to determine the impact of fixed vehicle rotations about one Orbiter axis with the hope of optimizing coverage. This approach is definitely feasible and does have a net positive effect in increasing antenna coverage, and can be used effectively to avoid long communication gaps. This approach, however, does assume that instruments which require specific fixed attitudes for target pointing are mounted on pointing platforms capable of offsetting the attitude corrections used to improve antenna coverage.

Conclusions - The analysis results presented for the AMPS mission simulation and a Z-LV mission attitude were factored into a detailed analysis of data retrieval for a fixed mission profile and experiment schedule, as illustrated in Figures 5.8.1-4 and -5. The data recovery profile shown in the upper portion of the figures indicates that use of only the primary K_u antenna is sufficient, when supplemented by

onboard recording, to recover data from the highest data volume experiment, minor constituents. Figure 5.8.1-4 shows TDRSS contact periods during the daily period of lower coverage indicated in Table 5.8.1-3. This period requires greater reliance on recording, with subsequent playback where feasible, or tape change. Figure 5.8.1-5 shows the period of higher TDRSS coverage, i.e., reduced antenna blockage, and less reliance on recording and tape change is obvious.

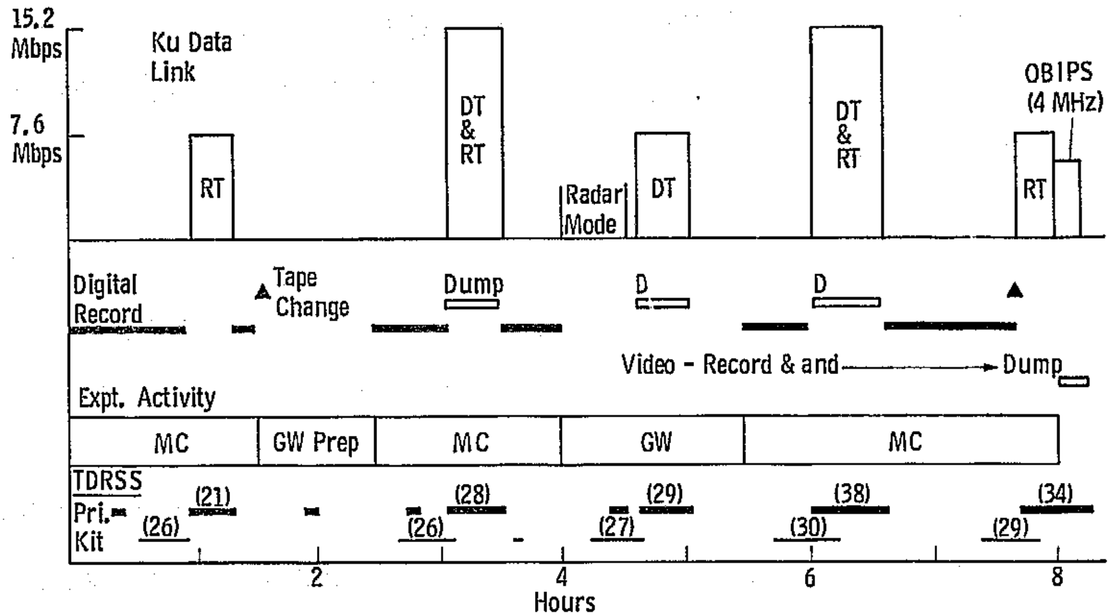


Figure 5.8.1-4 Data Recovery - Low TDRSS Coverage

Similar detailed data recovery analyses were not performed for Labcraft payloads requiring solar or stellar inertial attitudes. However, a comparison of the data handling requirements and the available coverages presented in Table 5.8.1-4 does allow for some conclusions in this area. Selection of the higher coverage antenna for the solar inertial attitude will probably accommodate the data recovery requirements. This may also be true for some stellar inertial missions, but an analysis of the specific attitude and mission requirements seems advisable to accurately predict antenna coverage.

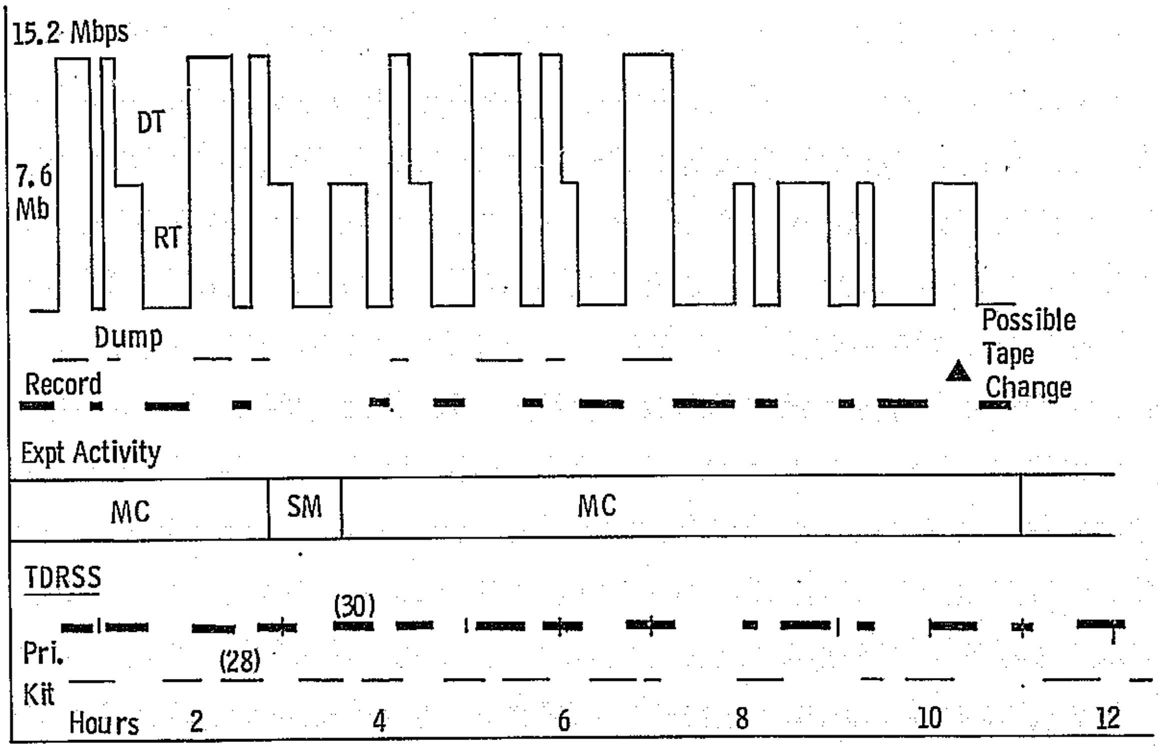


Figure 5.8.1-5 Data Recovery - High TDRSS Coverage

5.8.2 Communication Link Analyses

Analysis of the individual communication links to arrive at feasible designs and satisfactory circuit margins was the primary tool used for hardware selection. The links analyzed handle S-band communications with deployed packages or provide Ku-band tracking of those packages. Specifically not included in this analysis are the Orbiter-to-TDRS links which are considered operational facilities available for payload support. Following sections will describe the individual telemetry and command link analyses, and one section will summarize the tracking link analyses.

Gas/Chemical Release Analysis - Six gas release (GR) packages are ejected from the payload bay during Flight 1, and follow a trajectory which carries them about 80 Km from the Orbiter before the gas is rapidly released. Communication requirements consist of a relatively few commands required to arm the release mechanism in mid-flight and trigger the gas release at the appropriate point in space.

Since this requirement is functionally compatible with the Orbiter detached payload links, our initial design utilized this capability for S-band command transmission, and is presented in Table 5.8.2-1.

Table 5.8.2-1 Gas Release (GR) Command Link (Orbiter Terminal)

Orbiter EIRP	+ 1 dBw	Per JSC 07700, Vol XIV
Propagation Loss	-137.6 dB	2250 MHz, 80 Km
Polarization Loss	- 3 dB	Circular/Linear
GR Antenna Gain	- 3 db	Stub Antenna
GR RF Losses	- 1 dB	
Received Power, P_R	-143.6 dBw	
GR KTB	-153.8 dBw	Assume $T=600^{\circ}K$ and B=50 KHz
P_R/N	+ 10.2 dB	

Receiver Sensitivity, Rs=K Te B, Te=To (NF-1)

$$(NF-1) = Rs/K\ To\ B$$
$$= -153.8\ dBw - [-228.6 + 24.6 + 47]\ dBw$$
$$= -153.8\ dBw + 157\ dBw$$

NF = 2.09 + 1 = 3.09 or 4.9 dB

Two items in this table are worthy of note. First of all the resulting P_R/N is only about 10 dB, whereas a +12 dB is desirable at this point since we are dealing with a standard FM link and its associated receiver FM threshold. However, the 1? dB could be obtained by using a conical spiral rather than stub antenna on the GR. The second point, however, is a much more important factor; and it deals with the GR receiver sensitivity required to operate with the Orbiter signal. In order to achieve a P_R/N of 10 dB with the indicated P_R, a receiver sensitivity of about -153.8 dBw is required which equates to a receiver NF of under 5 dB. These characteristics are only obtainable with a very high quality receiver supplemented, possibly, with front end cooling. As a means of comparison, the near earth NASA standard transponder receiver has an NF of 6.5 dB and a sensitivity of -151.6 dBw.

For multiple, nonretrievable packages, use of such a high quality receiver was not considered to be cost effective. With an approximate 10 dB increase in radiated power, more typical, and less costly components could be used on the GR. The design for such a link is presented in Table 5.8.2-2. A commercially available, low cost FM receiver NF is used as the starting point, from which a minimum P_R is calculated and, subsequently, the minimum required AMPS EIRP. This design is also applicable to the Flight 2 chemical release.

Table 5.8.2-2 GR Command Link AMPS Terminal

Receiver NF	12 db
$R_S = KTeB = K\ T_o\ B\ (NF-1)$	
$= -157$ dBw $(15.85-1)$	
$= -157$ dBw $+ 11.7$	
$R_S = -145.3$ dBw	
Desired P_R/N	+ 12 dB
Required P_R	-133.3 dBw
Propagation Losses	-144.6 dBw (see Table 5.8.2-1)
Required AMPS EIRP	+ 11.3 dBw
AMPS RF Loss	- 0.5 dB
AMPS Antenna Gain*	+ 1 dBi (conical spiral antenna)
AMPS P_o	+ 10.8 dBw = 12 watts

*Close to peak antenna gain is achievable by Orbiter attitude correction.

ESP Link Analysis - The ESP maps the Orbiter bay while attached to the RMS and is then ejected, and follows a trajectory which departs no more than 4 Km from the Orbiter. Operation of the package requires command transmission as well as the recovery of 16 Kbps of telemetry. This requirement is compatible with the Orbiter or with the AMPS RF terminal. Both the link designs are included in Tables 5.8.2-3 and 5.8.2-4. As both these tables show, either the Orbiter or an AMPS terminal will operate compatibly with the ESP.

Table 5.8.2-3 ESP Command Link

	Orbiter		AMPS Terminal
Command P_o	---		+ 10 dBw
Command Antenna Gain	---		+ 1 dBi
Command RF Losses	---		- 1 dB
Command EIRP	+ 1 dBw		+ 10 dBw
Propagation Loss (2250 MHz, 4 Km)		-111.5 dB	
Polarization Loss (circular pol. antenna)		- 1 dB	
ESP Antenna Gain (conical spiral)		- 1 dBi	
ESP RF Loss		- 0.5 dB	
ESP P_R	-113.0 dBw		-104.0 dBw
ESP R_s (NF=12 dB)		-141.8 dBw	
P_R/N	- 28.8 dB		+ 37.8 dB
P_R/N Required		+ 12 dB	

RF Receiver Link Analysis - The RF Receiver package is deployed from the payload bay a distance of about 80 Km to conduct propagation and sounding experiments. The package requires tracking via the Orbiter, an S-band command link and the retrieval of a combination of digital (4 Kbps) and analog (30 KHz) data on a single RF carrier. The returned data will carry the analog on baseband, and use a subcarrier for the digital data, all of which will require an RF bandwidth of about 125 KHz. Since the Orbiter detached payload receiver can only accommodate digital data, a dedicated AMPS receiver is required. The command requirement is very similar to that of the Gas Release, and for the same reasons will utilize the AMPS command transmitter. There

Table 5.8.2-4 ESP Telemetry Link

	Orbiter		AMPS Receiver
ESP P_o	+ 3 dBw		0 dBw
ESP Antenna*	- 1 dBi		- 1 dBi
ESP RF Loss*	- 3.5 dB		- 3.5 dB
ESP EIRP	- 1.5 dBw		- 4.5 dBw
Propagation Loss (2250 MHz, 4 Km)		-111.5 dB	
Polarization Loss		- 1 dB	
Receiver Antenna Gain	- 1 dBi		- 1 dBi
Receiver RF Loss	- 3.5 dB		- 1 dB
P_R	-118.5 dBw		-119 dBw
R_s (NF=12 dB)		-142.3 dBw	
P_R/N	+ 23.8 dB		+ 23.3 dB

*P_o is split between dual antennas

are only minor differences in the command link design as shown in Table 5.8.2-5; and the proposed telemetry link design is presented in Table 5.8.2-6. The telemetry design includes a 10 watt power output split between 2 conical spiral antennas mounted on opposite faces of the package. It should be noticed that a lower noise figure receiver was necessary, or a low noise preamplifier, to obtain the minimum acceptable +12 dB P_R/N.

RMS Packages Link Analysis - A trade study was performed to determine the technique for the transfer of commands and data between packages deployed on the RMS and the Spacelab data system. Because the RF approach was less complex, involved much less hardware development, and was lower in cost, this technique was selected over a hardwire approach requiring rather unique cable deployment mechanisms. The instrument packages included in this study were the Flight 1 ESP and Beam Diagnostics package and the Flight 2 Plasma Wake Generator and Diagnostics packages. There is no difficulty in achieving strong circuit margins since the propagation path is no more than 20 meters. Problems are more likely to be encountered in the areas of near-field propagation and multipath. A typical downlink case is analyzed in Table 5.8.2-7 for the wide band data requirement associated with the Beam Diagnostics package. The other links will be similar in design, with lower bandwidth requirements. For the command links, it is

Table 5.8.2-5 RF Receiver Command Link

AMPS Command P_o	+ 10 dBw
AMPS Antenna Gain	+ 1 dBi
AMPS RF Loss	- 1 dB
AMPS EIRP	+ 10 dBw
Propagation Loss (2250 MHz, 80 Km)	-137.6 dB
Polarization Loss (circular/circular)	- 1 dB
RF Receiver Antenna Gain	- 1 dBi
RF Receiver Loss	- 3.5 dB
P_R	-133.1 dBw
Receiver Sensitivity (NF=12 dB, T=600° K, B=25 KHz)	-148.3 dBw
P_R/N	+ 15.2 dB

Table 5.8.2-6 RF Receiver Telemetry Link

RF Receiver P_o	+ 10 dBw
RF Receiver Antenna Gain	- 1 dBi
RF Receiver RF Losses *	- 3.5 dB
RF Receiver EIRP	+ 5.5 dB
Propagation Loss (2250 MHz, 80 Km)	-137.6 dB
Polarization Loss	- 0.5 dB
AMPS Antenna Gain	+ 1 dBi
AMPS RF Losses	- 1 dB
AMPS P_R	-132.6 dBw
AMPS Receiver Sensitivity (T=600° K, B=125 KHz, NF=8 dB)	-145.0 dB
P_R/N	+ 12.4 dB

*P_o is split between dual antennas

Table 5.8.2-7 RMS Wideband Link

RMS Package P_o	- 10 dBw	(100 m watts)
RMS Package Antenna Gain	- 3 dBi	
RMS Package RF Loss	- 1 dB	
RMS Package EIRP	- 14 dBw	
Propagation Loss (2250 MHz, 60 ft)	- 65 dB	
Polarization Loss	- 3 dB	
AMPS Antenna Gain	+ 1 dBi	
AMPS RF Loss	- 1 dB	
AMPS P_R	- 82 dBw	
AMPS Receiver Sensitivity ($T=600^{o}$ K, B=5 MHz, NF=15)	-122 dBw	
P_R/N	+ 40 dB	

possible that receiver saturation will occur if the AMPS command transmitter 10 watt output is not attenuated for these RMS applications. For example a command transmission attenuated by 20 dB will provide a circuit margin of nearly 50 dB.

<u>S-Band Link Analysis Summary</u> - The link analysis results presented in this section are summarized in Figure 5.8.2-1.

Link Parameter	Gas/Chemical Releases	ESP	RF Receiver Package	RMS Instruments
Range	80 km	4 km	80 km	<25 m
Telemetry				
Data	Not Required	16 kbps	60 kHz (Digital + Analog)	4 MHz (Flight 1) 100 kHz (Flight 2)
Power		1 W	5W	<0.1 W
Carrier/Noise		>20 dB	12 dB	>30 dB
Command				
Power (AMPS)	12 W	10 W	10 W	< 10 W
Modulation	Tones	Digital	Digital	Digital
Carrier/Noise	12 dB	>25 dB	15 dB	>30 dB

Figure 5.8.2-1 S-band Link Analysis Summary

K_u-Band Tracking Link Analysis - The Orbiter K_u radar is used to skin track the instrument packages deployed from the payload bay. Specific characteristics of the Orbiter system were extracted from the "Shuttle Orbiter Ku Band Radar/Communications Subsystem" report presented at the August 13, 1975, bidders' seminar by Rockwell International. Tracking link analyses were performed for each of the deployed packages using the minimum cross-sectional area of each package. One typical link analysis is presented in Table 5.8.2-8 for the smallest package, the RF Receiver. The analysis considers a minimum desired tracking range of 10 Km since the tracking data is to be used to update the predicted trajectory of the deployed package. Tracking beyond this range is not necessary. The radar equation used was:

$$P_R/N_o \text{ (Ave)} = \frac{P_o \, G^2 \, \lambda^2 \, \delta \, T_d}{(4\pi)^3 \, R^4 \, K \, T \, L \, (NF)}$$

where the parameters are defined in Table 5.8.2-8.

Table 5.8.2-8 K_u-Band Tracking Link

Orbiter Radar P_o (20 watts ave)	+ 13 dBw	
Orbiter Antenna Gain, G^2 (X2)	+ 70.8 dB	
Wavelength, λ^2 ($2\text{x}10^{-2}$ m)		- 34 dB
Radar Cross Section, δ (0.22 m^2)		- 6.6 dB
Dwell Time, T_D (80 ms)		- 11 dB
$(4\pi)^3$		- 33 dB
Range, R^4 (10 Km)		-160 dB
Receiver Noise Figure, NF=8 dB		- 8 dB
K ($1.38\text{x}10^{-23}$)	+228.6 dB	
Temperature, T (290° K)		- 24.6 dB
Losses, L (from reference report)		- 17.9 dB
Totals	312.4 dBw	-295.1 dB
	-295.1 dB	
P_R/N_o	+ 17.3 dB	
Minimum Required P_R/N_o	+ 15.0 dB	

The average carrier-to-noise ratios for the other deployed packages are tabulated in Figure 5.8.2-2, which also contains the package cross sectional areas and the maximum tracking range possible.

Link Parameter	Releases		ESP	RF Receiver
	Gas	Chemical		
Minimum Cross Section	0.7 m^2	1.5 m^2	0.48 m^2	0.22 m^2
Maximum Tracking Range°	15 km	18 km	4.0 km°°	11.2 km
Carrier/Noise at Range = 10.0 km	22 dB	25 dB	28 dB (at 4.0 km)	17 dB

° Results in minimum 15-dB carrier/noise.

°° Maximum range from orbiter.

Figure 5.8.2-2 Radar Tracking Analysis

5.8.3 Flux Density Analysis

International regulations exist which are intended as a control on RF radiation from space vehicles, particularly for those situations where the radiation is pointed in the general Earth direction. The concern, obviously, is that these radiations may interfere with terrestrial communication links. These regulations limit the power flux density (PFD), i.e., concentration of RF power in a specified bandwidth of 4 KHz, depending on the signal's angle of incidence with the Earth. The nature of this constraint is illustrated in Figure 5.8.3-1 which specifies the following limitations on PFD for S-band radiations:

PFD $< -$ 144 dBw for $\theta \geq 25^{\circ}$

PFD $< -$ 154 dBw for $0^{\circ} < \theta < 5^{\circ}$

PFD $< -$ (154 + n) dBw for $\theta = 5 + 2n^{\circ}$ (up to 25°)

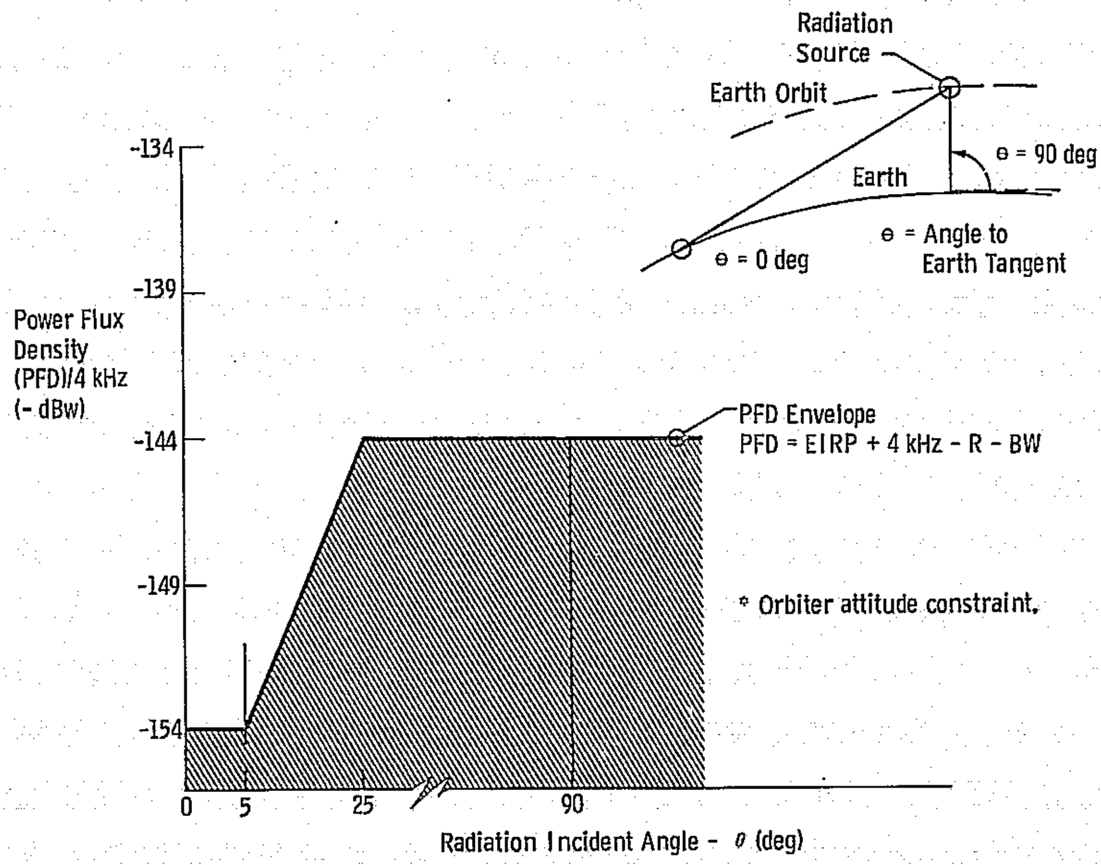


Figure 5.8.3-1 Power Flux Density Profile

The PFD limitation is lower for small angles of incidence because these signals are more likely to fall within terrestrial antenna patterns and offer the greater probability of interference.

It should be noted at this point that these regulations were not identified as a specification on the design of AMPS RF links. It seemed reasonable, however, to use these regulations as guidelines and to steer the RF design in this direction.

Figure 5.8.3-1 also shows that PFD increases with source radiated power and is decreased by increasing data rates or RF bandwidth, thus suggesting that the radiated signal be spread over a large enough bandwidth to satisfy the PFD limitations.

All of the radiation sources directly related to the AMPS payload were analyzed to determine their compatibility with these PFD limitations and reasonable adjustments were made to approach these limitations, which are described in the following sections.

Analysis of EMI Package - The initial design of the EMI telemetry link required an EIRP of - 1 dBw and BW of 50 KHz. For these parameters,

$$\text{PFD at } \theta = 90^{o} = -1 \text{ dBw} + 10 \log (4 \text{ KHz}) - 10 \log (4 \pi R^{2}) - 10 \log (50 \text{ KHz})$$

where R = 210 Km range at $\theta = 90^{o}$

$$\text{PFD } (\theta = 90^{o}) = -1 \text{ dBw} + 36 \text{ dB} - 117.2 \text{ dB} - 47 \text{ dB}$$
$$-129.3 \text{ dBw}$$

which is over 10 dB above the PFD limit.

Bandwidth spreading of this signal to satisfy the - 144 dBw limit value would require:

$$\text{RF BW} = \text{EIRP} + 4 \text{ KHz} - \text{Range} - \text{PFD}$$
$$= -1 \text{ dBw} + 36 \text{ dB} - 117.2 \text{ dB} + 144 \text{ dBw}$$
$$= 61.7 \text{ dB or nearly } 1.5 \text{ MHz}$$

Spreading a basic 16 Kbps telemetry signal across 1.5 MHz requires sophisticated and costly modulation techniques not compatible with a low cost, nonretrievable instrument package. The only remaining, flexible parameter is the EIRP which was primarily fixed by the Orbiter receiving characteristics associated with the detached payload links. These characteristics included an omni-directional, low gain antenna and a receiver sensitivity of - 133.8 dBw. The receiver sensitivity was recognized as a limiting factor in reducing the EMI radiated power. Consideration was then given to using an alternate receiver, probably payload dedicated hardware. As indicated in Section 5.8.2, the final link design

interfaced with a dedicated AMPS RF terminal containing a receiver with a - 146 dBw sensitivity, which enabled a reduction in the EMI package EIRP to - 7 dBw. The current PFD, therefore, at $\theta = 90^{o}$

$$= -7 \text{ dBw} + 36 \text{ dB} - 117.2 \text{ dB} - 48.1 \text{ dB} \quad \text{(bandwidth = 65 KHz)}$$

$$= -136.3 \text{ dBw}$$

for $\theta = 0\text{-}5^{o}$,

$$\text{PFD} = -7 \text{ dBw} + 36 \text{ dB} - 10 \log (4\pi \times [1600 \text{ Km}]^2) - 48.1 \text{ dB}$$

$$= -154.5 \text{ dBw}$$

Analysis of RF Receiver Package - Early estimates of the RF Receiver package power output were in the range of 10-20 watts, which would result in a PFD of about - 120 dBw. However, by selecting a lower noise figure receiver, it was possible to reduce the radiated power level to the point where:

$$\text{PFD} (\theta = 90^{o}) = +5 \text{ dBw} + 36 \text{ dB} - 117.2 \text{ dB} - 51.8 \text{ dB}$$

(for a 150 KHz bandwidth)

$$= -128.1 \text{ dBw}$$

and $\text{PFD} (\theta = 0\text{-}5^{o}) = -146.3$ dBw

Both of these levels are well above the PFD limits presented earlier, and the following options are available for improving the PFD levels:

a - increase radiated signal bandwidth

b - use lower noise figure receiver or pre-amplifier

c - use a higher gain receiving antenna.

Although the signal bandwidth can be increased by conventional modulation or spread spectrum techniques, this would impact the link circuit margin unacceptably. This approach would be more feasible if accompanied by the use of a lower noise figure receiver, but this combined change is not the cost effective approach, and still would not achieve the desired PFD limits.

The preferred option for significantly reducing the PFD levels would be the use of a higher gain receiving antenna, i.e., replacing the proposed conical spiral with a more directional horn antenna, which could increase the gain by about 10 to 15 dB. Since this antenna satisfies a number of RF requirements, the impact on the other applications would require analysis before the antenna change is made, assuming that satisfying the flux density levels does become a program requirement.

Analysis of Command Transmissions - Several packages deployed from the Orbiter bay require the transmission of commands via RF link. Two sources of these commands were investigated to determine PFD levels, namely, the Orbiter detached payload transmitter and a dedicated AMPS command transmitter. The Orbiter transmit characteristics are fixed, leaving no room for trade-offs to improve link parameters and resulting in the following:

$$PFD\ (\theta = 90^{o}) = +1\text{ dBw} + 36\text{ dB} - 117.2\text{ dB} - 47\text{ dB}$$ (Assuming a nominal command bandwidth of 50 KHz)

$$= -127.5\text{ dBw}$$

The other option for command control of deployed instruments utilizes the AMPS terminal described in Section 4.7, which consists of a 10 watt command transmitter and results in the following:

$$PFD\ (\theta = 90^{o}) = +10\text{ dBw} + 36\text{ dB} - 117.2\text{ dB} - 47\text{ dB}$$
$$= -118.3\text{ dBw}$$

Obviously, these results are well out-of-spec, but they are presented merely to give as complete a picture as possible. No situation has been identified during the flights investigated which would require command transmissions pointed toward the Earth. On the contrary, these transmissions will either be pointed away from Earth or along the orbital path to a trailing deployed instrument. In this latter case, however, it is possible that sidelobe energy would impinge on the Earth. Figure 5.8.3-2 shows, for example, that radiation directed toward a trailing object will impinge on the Earth at about 25 degrees or more off the main antenna axis. For such a case:

$$PFD\ (\theta = 0\text{-}5^{o}) = 2\text{ dBw} + 36\text{ dB} - 135.4\text{ dB} - 47\text{ dB}$$
$$= -148.4\text{ dBw}$$

This calculation considers a 3 dB reduction in Orbiter EIRP since we are operating below peak antenna gain. The command antenna proposed for the dedicated AMPS RF terminal has a wider antenna pattern, and the possibility of impingement is greater. Considering a likely 3 dB reduction in EIRP for operation 25 degrees off the antenna axis:

$$PFD\ (\theta = 0\text{-}5^{o}) = +7\text{ dBw} + 36\text{ dB} - 135.4\text{ dB} - 47\text{ dB}$$
$$= -139.4\text{ dBw}$$

Impingement at higher angles is possible, but the reduction in EIRP will probably cancel out the reduction in range loss, resulting in little change to the PFD level. It is possible that further reduction in these levels could be achieved either by Orbiter attitude correction to further reduce radiation in the Earth's direction, or by consideration of a more directional antenna such as a horn.

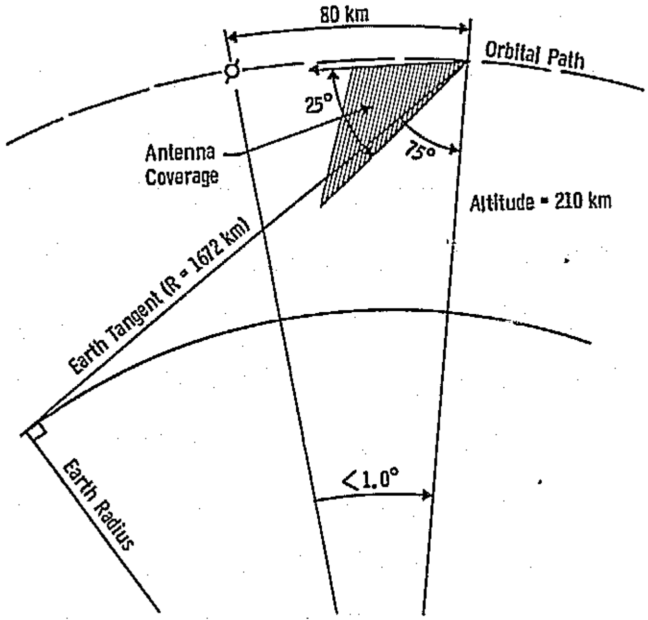


Figure 5.8.3-2 Command Radiation Geometry

Analysis of RMS Transmissions - Packages mounted on the RMS will radiate telemetry toward the Orbiter bay, and depending on altitude, it is possible that this energy could be directed toward the earth at angles as high as 90 degrees. For such a situation:

$$PFD\ (\theta = 90^{o}) = -\ 13.5\ dBw + 36\ dB - 117.2\ dB - 47\ dB$$

$$= -141.8\ dBw$$ for the minimum bandwidth case of 50 KHz

and for the wide bandwidth (4 MHz) case:

$$PFD\ (\ \theta\ =\ 90^{o}) = -\ 160.85\ dBw$$

Because of the low radiation levels, these links should not cause terrestrial interference problems.

Flux Density Analysis Summary - A summary of the flux density analysis results is illustrated in Figure 5.8.3-3 with respect to the PFD limitation envelope. As already indicated in this section, a higher gain AMPS receiving antenna would significantly lower most out-of-spec levels. However, it should also be taken into consideration that all the transmitted signals discussed are of very short duration, as is the entire mission itself; and thus would not represent a permanent or long term source of terrestrial interference.

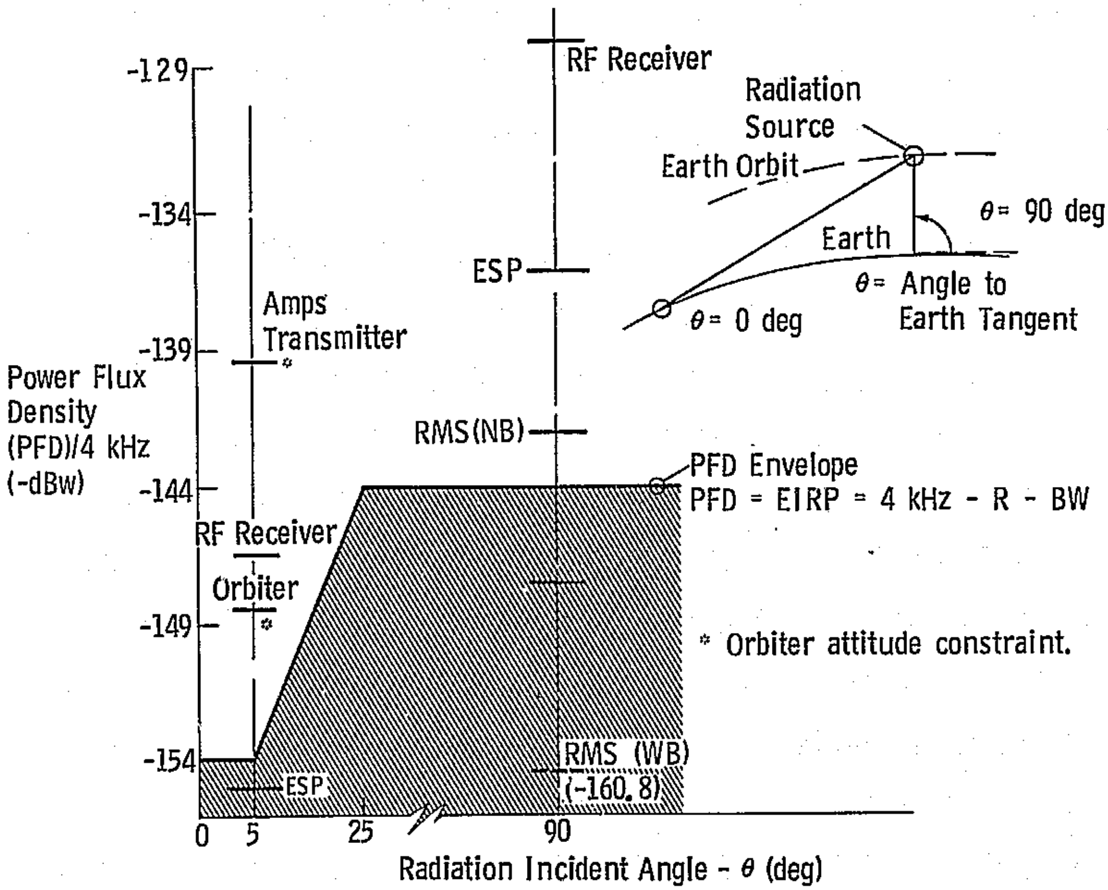


Figure 5.8.3-3 Flux Density Analysis Summary

5.9 Deployed Instrument Support

5.9.1 Early Maneuverable Subsatellite vs ESP and Other Free Flying Satellites

GSFC advance planning documentation (Strawman Payloads for First 5 AMPS Missions) for AMPS calls out a maneuverable subsatellite (MSS) for Flight 4 to support the deployed instrument experiments. The question we shall address in this section is: What are the scientific advantages and cost factors associated with the use of a maneuverable subsatellite in Flights 1 and 2 of AMPS as compared with free flying subsatellites? This is based on the assumption that an MSS will be funded so as to be available for AMPS Flight 4 and the question here is that of moving the MSS earlier in the AMPS program. This section discusses the AMPS requirements imposed on an MSS, and identifies some candidate MSS vehicles.

5.9.1.1 Scientific Enhancement of Experiments

Each of the deployed instrument experiments was examined for its possible scientific enhancement using an MSS. The following experiments fell into this category:

- o Electron Beam Studies, Level II
- o EMI Field Mapping and Orbiter Wake Measurements
- o Conductivity Modification
- o Long-Delay Echo and Wave/Particle Interactions
- o Plasma Flow/Wake Generator

Electron Beam Studies - The present implementation of the measurements on the Electron Beam Studies, Level II, as discussed in Sections 3.4 and 4.8, is limited to a near-field exploration using the Shuttle RMS. These near-field measurements of the electron beam are compromised by the closeness to the Orbiter and the presence of RMS.

The primary benefit from using an MSS on this experiment would be beam measurements and mapping at significantly further distances than those afforded by the RMS. The beam diagnostic (in-situ) sensors would be mounted on the MSS.

Figure 5.9.1-1 shows a typical scenario. The MSS carries the beam diagnostic package and moves from the Orbiter to a distance of about 500 meters and maps the electron beam. This is repeated twice bringing the MSS out to a distance of about 1500 meters. When we analyze this representative MSS path, and allow 30 minutes to move each of the 500 meters, we obtain a ΔV requirement of 16 m/s.

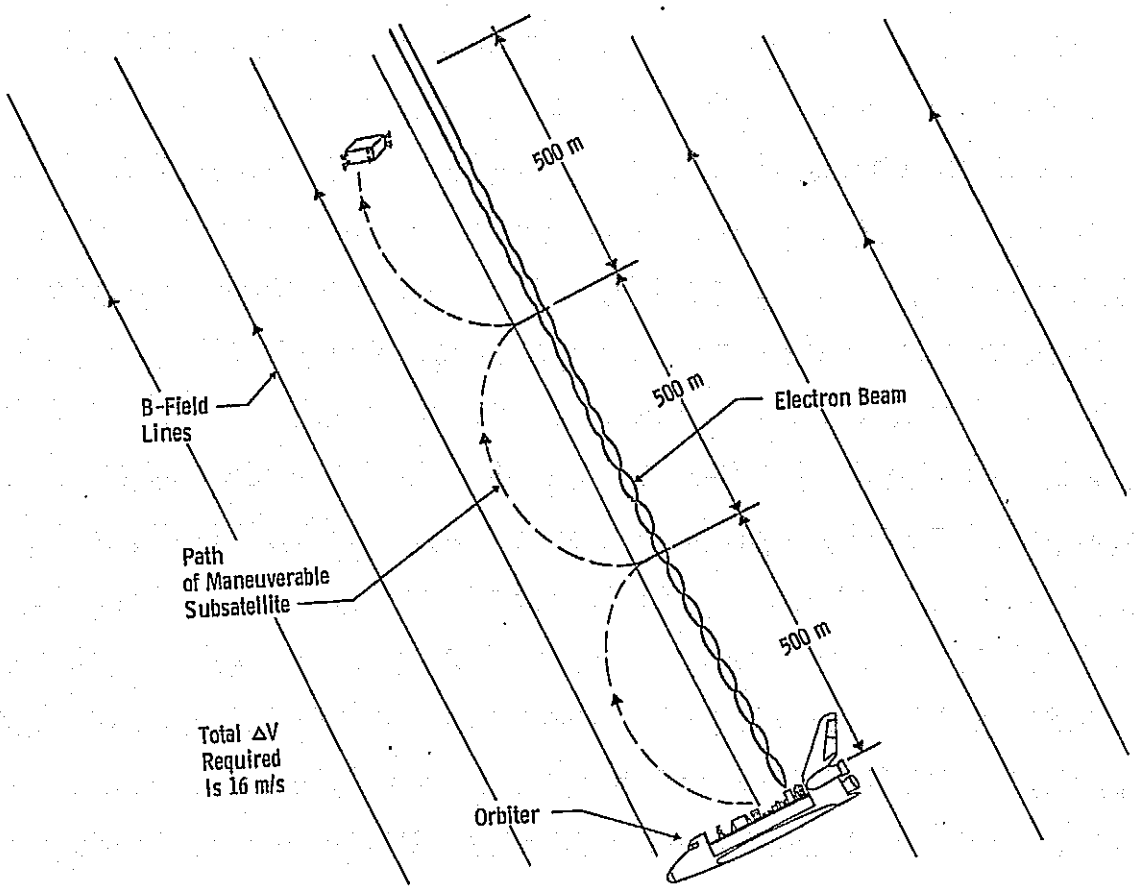


Figure 5.9.1-1 Electron Beam Studies, Level II Diagnostics Using a Maneuverable Subsatellite

<u>EMI Field Mapping and Orbiter Wake Measurements</u> - The present baseline measurement is limited to a near-field exploration (RMS range of access), plus a one-time traversal of the far-field by the ejected ESP package. The near-field measurements are compromised by the closeness to the sources and by the presence of the RMS. The far-field measurements of EMI and wake measurement are limited in that only one ambient plasma condition can be explored and the single traversal limits the number of operating configurations for which EMI signatures can be obtained. The wake traversal is limited to the ESP trajectory which is not necessarily at the preferred distance from the wake generator (Orbiter). Here again the wake characteristics, as a function of changing ambient plasma conditions (i.e., daylight, dark, high latitude, mid-latitude, South Atlantic anomaly, etc.) and

varying the attitude of the Orbiter, would greatly increase the understanding of the wake structure.

Figure 5.9.1-2 depicts a typical mission using an MSS to explore the Orbiter wake. The MSS carries the ESP instruments back and forth across the wake, firing its thrusters such that it coasts through the wake. The crossings are calculated to take 5 minutes. The total ΔV required to complete this mission is 18 m/s.

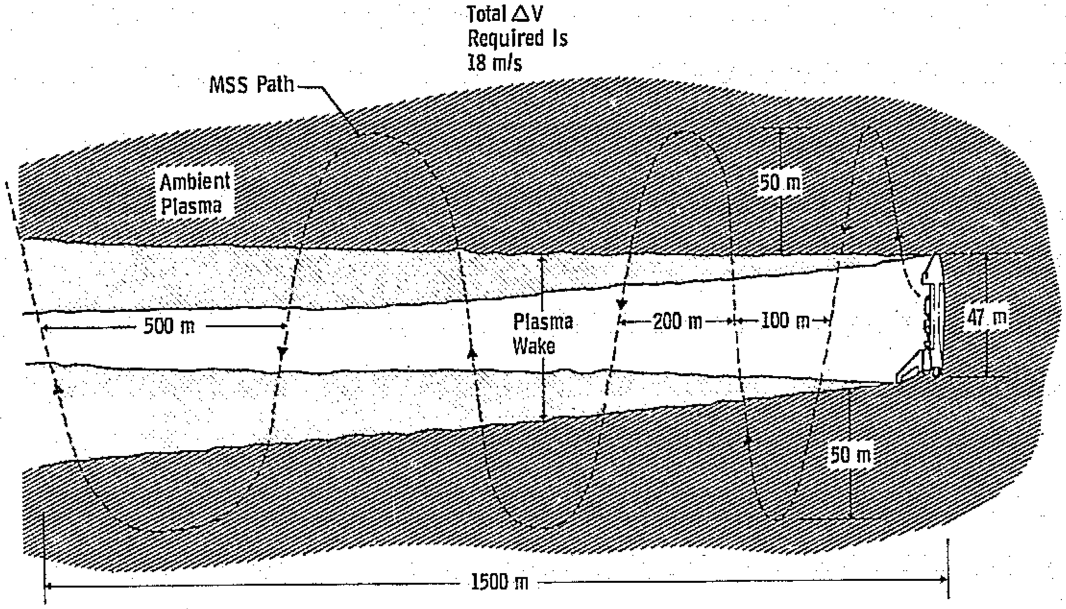


Figure 5.9.1-2 Orbiter Wake and EMI Far-Field Measurements Using Maneuverable Subsatellite

<u>Conductivity Modification</u> - This experiment can be enhanced by using a MSS to carry instruments through or near the Barium cloud 10 to 15 minutes after the cloud is released (see Figure 5.9.1-3. This provides important observations, supplementing those taken from the ground and from the Orbiter. However, to place the MSS 10 to 15 minutes (6000 Km) behind the Orbiter prior to the chemical module deployment, requires a ΔV of 70 m/s. Then to retrieve the MSS we use another 140 m/s giving a ΔV requirement of 220 m/s. This includes 10 m/s for maneuvering in the vicinity of the Orbiter.

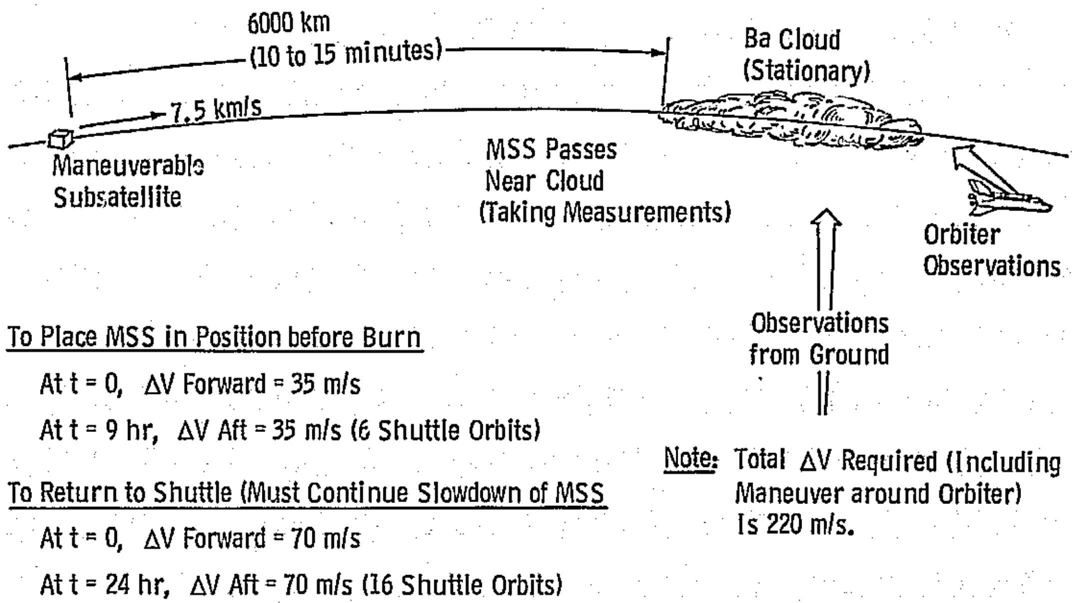


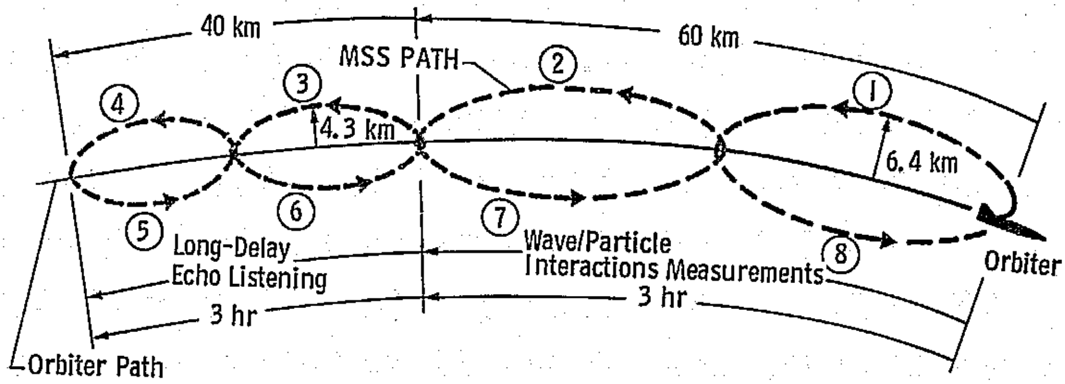
Figure 5.9.1-3 Conductivity Modification Using an MSS

The high ΔV, coupled with the limited scientific enhancement using an MSS for this experiment, places its recommendation in doubt.

Long-Delay Echo and Wave/Particle Interactions - The limitations of the baseline implementation is the one-time traversal of the zone of interest. The MSS would allow us to explore a wide variety of magnetospheric latitudes and locations by means of a receiver on controllable vehicle. In addition, the ability to control the relative orientations of the transmitter and receiver antennas and the orientation with respect to the $\overline{B}$ field vector would greatly enhance the experiment.

Figure 5.9.1-4 shows a typical trajectory for an MSS while measuring the long delay echo and wave/particle interactions. The wave/particle interactions measurements are taken during the first three-hour, two-orbit, period. This is shown by numbers 1 and 2 in the figure. This brings the MSS to a point 60 km behind the Orbiter. Starting from this point, the instruments on the MSS begin listening for the long-delay echo. For purposes of estimating the ΔV required, we have shown a trajectory where 40 km is covered during a three-hour period, bringing the MSS to a point 100 km behind the Orbiter. (Orbits 3 and 4 in the figure.) Then a return course (orbits 5 through 8) is assumed to be the mirror image of the "going" trajectory. In actuality, the MSS would be maintained in a fixed position, relative to the Orbiter, where the long-delay echo appears to have

a maximum signal strength. The total ΔV required for this experiment is 17.6 m/s.



From 0 to 60 km in 2 Orbits: ΔV = 1.9 M/s Forward
From 60 to 100 km in 2 Orbits: ΔV = 0.6 M/s AFT
From 100 to 60 km in 2 Orbits: ΔV = 2.6 M/s AFT
From 60 to 0 km in 2 Orbits: ΔV = 0.6 M/s AFT
At Orbiter = 1.9 M/s Forward

Total ΔV (Including Maneuvering around Orbiter) = 17.6 M/s

Figure 5.9.1-4 Long Delay Echo and Wave/Particle Interactions Using a Maneuverable Subsatellite

Plasma Flow Studies - This experiment can be enhanced by the use of an MSS. The present implementation is significantly limited by the reach of the Shuttle RMS. In addition, the wake from the balloon can be influenced by the proximity of the Orbiter and the RMS-mounted extension boom.

To perform this experiment, using an MSS, we release the balloon from the RMS and map its wake using MSS-mounted plasma wake diagnostic package. Figure 5.9.1-5 depicts this experiment with the MSS crossing the wake at increasing distances from the balloon. This approach provides a more complete wake mapping and provides more accurate data. Allowing three minutes for each traversal across the wake, we obtain a total ΔV requirement of 12.7 m/s.

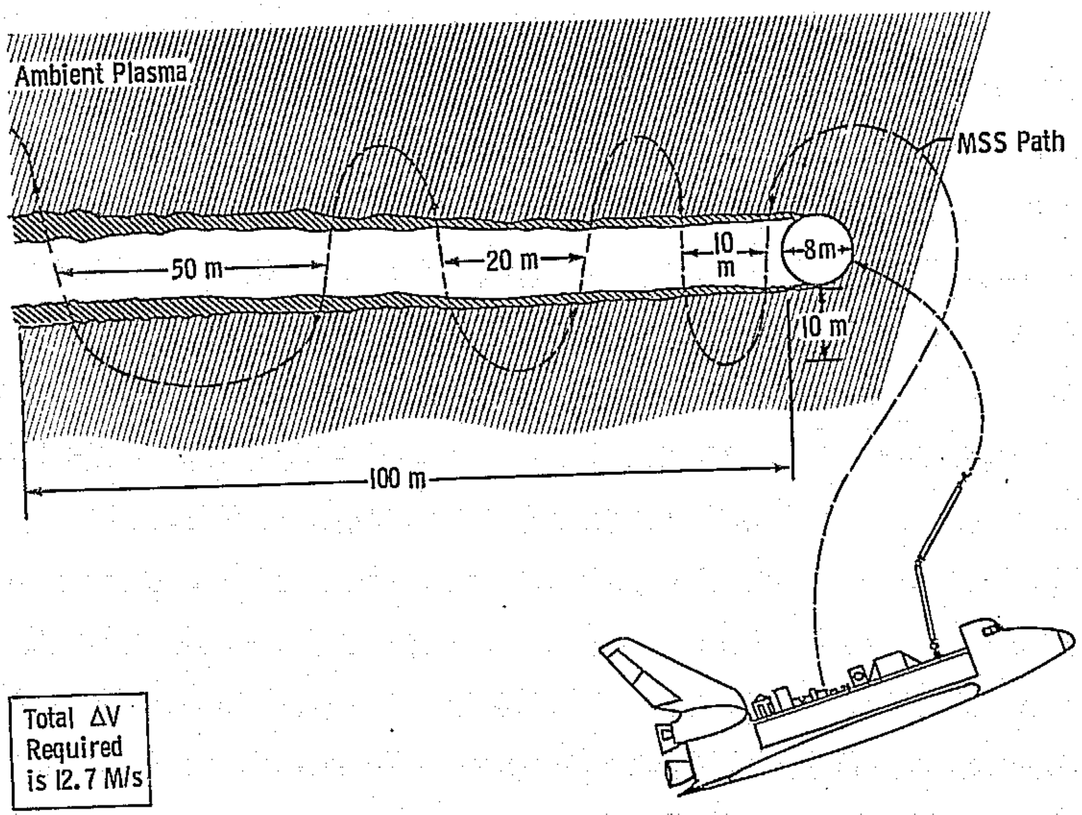


Figure 5.9.1-5 Plasma Flow/Wake Generator Experiment Using an MSS

5.9.1.2 AMPS Maneuverable Subsatellite Selected Requirements

Table 5.9.1-1 is a summary of the requirements that each of the above described experiments places on the MSS. The impulse shown in the table was calculated based on a vehicle mass of 240 kg. The impulse required would change, of course, as different vehicle masses were used. Also, the table includes the contamination concerns for each experiment.

Table 5.9.1-1 Summary of Selected MSS Requirements

Requirement / Experiment	Package Size	Package Mass, kg	ΔV, m/s	Typical Impulse (Based on Vehicle Mass of 240 kg), lb-s	Vehicle Contamination Concerns
Electron Beam Studies, Level II Diagnostics	Rectangular 15x18x36 in.	115	16	1166	Magnetic Field
Orbiter Wake and EMI Far-Field	Cylindrical 18 in. Deep 41 in. Dia	140	18.4	1435	Magnetic Field Thrusters
Conductivity Modification	Cylindrical 18 in. Deep 41 in. Dia	140	220	17,160	Magnetic Field Thrusters
Long-Delay Echo and Wave/ Particle Interactions	Cylindrical 16 in. Deep 21 in. Dia	39	17.6	1007	Magnetic Field
Plasma Flow	Rectangular 15x18x36 in.	58	12.7	777	Magnetic Field Thrusters

5.9.1.3 AMPS Maneuverable Subsatellite Candidates

We examined an array of spacecraft as possible candidates for an AMPS MSS. We included spacecraft that are presently either operational, under development or in the conceptual phase. Seven spacecraft appear to be feasible candidates for an AMPS MSS. Table 5.9.1-2 lists these spacecraft, their status, and the modifications required to meet the AMPS MSS requirements. All of the spacecraft reviewed would require some degree of modification to the basic subsystems and/or the addition of subsystems. The modifications to the basic spacecraft range from the relatively simple task of providing Shuttle and experiment interface adapters to what would amount to a total redesign, requiring such modifications as replacing solar panels, providing a three-axis stabilization system, or replacing a bipropellant system with a monopropellant or cold gas system.

All of the spacecraft listed in Table 5.9.1-2 are well documented except the newly conceived MTS (last entry in table.)

Table 5.9.1-2 MSS Spacecraft Candidates

Spacecraft	Status	Modifications Required
Atmospheric Explorer (AE)	Operational	Orbiter and Experiment Interfaces Collision Avoidance System Attitude and Translation Control
Synchronous Meteorological Satellite (SMS)	Operational	Orbiter and Experiment Interfaces Increase Propulsion Capability Collision Avoidance System
Earth Resources Technology Satellite (ERTS)	Operational	Orbiter and Experiment Interfaces Increase Propulsion Capability Collision Avoidance System Remove Solar Panels
Multimission Spacecraft (MMS)	Development Phase	Orbiter and Experiment Interfaces Collision Avoidance System Remove Solar Panels
Earth-Orbital Teleoperator System (EOTS)	Conceptual Phase	Experiment Interfaces
Space Test Program Standard Satellite (STPSS)	Conceptual Phase	Orbiter and Experiment Interfaces Collision Avoidance System Remove Solar Panels
Maneuverable Television System (MTS)	Conceptual Phase	Experiment Interfaces

The MTS was formerly the Manned Maneuvering Unit (MMU) designed to be operated remotely, unmanned. The MMU, an outgrowth of the Astronaut Maneuvering Unit, may be provided for early Shuttle flights and could possibly be considered for early use on AMPS. The MTS is a low cost remote control free-flying platform. It will be deployed from a cargo bay pallet for missions up to 3 hours in duration and operations out to approximately one mile. Control is provided from a portable command station at the Orbiter aft flight deck. Operator commands will be encoded and transmitted to the flyer for six degree of freedom control and camera adjustment. Telemetry and video data will be displayed on the Shuttle CCTV monitor for optimum man/machine interaction. Since the maneuvering platform portion of the

flyer is being derived from the astronaut Manned Maneuvering Unit (MMU) design, use will be made of MMU power, gyro electronics, control logic, propulsion configuration and mechanical latch designs to maximize commonality and minimize cost and risk. The MMU rate gyros and low force GN_2 thrusters provide a low rate stable attitude control mode for television viewing. The MMU control logic and thruster geometry will permit a large dispersion in CG location while retaining its propellant conservative characteristics. The telecommunications portion of the flyer will be made up of off-the-shelf telemetry, command and television subsystems. The TV camera is the key element for control of the man/machine system. It serves as the primary feedback sensor for remote control rotation and positioning of the flyer. The telemetry and TV video data are interleaved and downlinked by a single transmitter. Received uplink commands will be decoded and conditioned as necessary for MTV control.

5.9.1.4 Cost Aspects in the Use of a Maneuverable Subsatellite

There are several types of cost savings associated with the early use of an MSS for AMPS. These are, (1) instruments that are retrieved and are used on a later flight, (2) instruments that are retrieved without a specific planned reuse, (3) flight support equipment (such as ejection mechanisms) not required, and (4) Shuttle support equipment not required. These are discussed below.

The instruments used on Flight 1, Orbiter Wake and EMI Far-Field experiment uses the following instruments which can be reused in Flight 2.

Langmuir Probe
Planar RPA
Neutral Mass Spectrometer

This results in a savings of $0.210 Million.

Flight support equipment savings include communications, data management, electrical, mechanisms and structures for both Flights 1 and 2. These are approximately $2.730 million and $0.876 million respectively.

A requirement for the second Shuttle RMS can be eliminated if an MSS is used. This cost savings is not clear at this time.

The prime disadvantage in the proposed early use of an MSS is the requirement for early NASA funding to either modify an existing vehicle or develop a new vehicle.

6.0 LIST OF ACRONYMS

ADC (A/D)	Analog to Digital Conversion
ADS	Aerospace Data System
AE	Atmospheric Explorer
AEC	Atomic Energy Commission
AMPS	Atmosphere, Magnetosphere, and Plasmas in Space
A/N	Alphanumeric
AO	Announcement of Oppurtunity
APCS	Attitude Pointing and Control Subsystem
ATE	Automatic Test Equipment
ATM	Apollo Telescope Mount
ATP	Authorization to Proceed
CAMAC	Computer Automated Measurement and Control
C&D	Control and Display
C&W	Caution and Warning
CCTV	Closed Circuit Television
CDMS	Command and Data Management Subsystem
CDR	Critical Design Review
CG	Center of Gravity
CM	Center of Mass
CMD	Command
C/O	Checkout
COCOA	Computer Oriented Communication Operational Analysis
CRT	Cathode Ray Tube
CSS	Core Segment Simulator
DAC (D/A)	Digital to Analog Conversion
DMS	Data Management Subsystem
DOMSAT	Domestic Satellite
DT	Delay Time
EDU	Electrical Distribution Unit
EGSE	Electrical Ground Support Equipment
EIRP	Effective Isotropic Radiated Power
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EOR	Experiment Operation Requirements
EOTS	Earth Orbital Teleoperator System
EPTB	Electrical Power Distribution Box
EPDS	Electrical Power and Distribution Subsystem
ESA	European Space Agency
ESP	Environmental Sensing Package
ETR	Eastern Test Range
FHST	Fixed Head Star Tracker
FKB	Function Keyboard
FM	Frequency Modulation
FMEA	Failure Mode Effects Analysis
FORMA	Fortran Matrix Analysis
FOV	Field of View
FSE	Flight Support Equipment

GFE	Government Furnished Equipment
GMT	Greenwich Mean Time
GN&C	Guidance, Navigation and Control
GPC	General Purpose Computer
GR	Gas Release
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
GW	Gravity Wave
Hx	Heat Exchange
IECM	Induced Environmental Contamination Monitor
IFRD	Instrument Functional Requirements Document
IMU	Inertial Measurement Unit
I/O	Input/Output
IPS	Instrument Pointing System
IR	Infrared
JSC	Johnson Space Center
KB	Keyboard
KSC	Kennedy Space Center
LIDAR	Light Detection and Ranging
LOS	Line of Sight
LPS	Launch Processing System
MC	Minor Constituent
MCC	Mission Control Center
MDM	Multiplexer/Demultiplexer
MET	Mission Elapsed Time
MITAS	Martin Interactive Thermal Analysis System
MLI	Multilayer Insulation
MMC	Martin Marietta Corporation
MMSE	Multi-use Mission Support Equipment
MMU	Manned Maneuvering Unit
MPM	Miniature Pointing Mount
MSS	Mission Specialist Station
MSFC	Marshall Space Flight Center
MTS	Maneuverable Television System
MUX	Multiplexer
NIM	Nuclear Instrument Module
OBIPS	Optical Band Imaging and Photometric System
O&C	Operation and Checkout
OIA	Orbiter Interface Adapter
OMS	Orbit Maneuvering System
OPF	Orbiter Processing System
PCM	Pulse Code Modulation
PCR	Payload Checkout Room
PDR	Preliminary Design Review
PFD	Power Flux Density
PHF	Payload Handling Facility
PIU	Pyro Initiator Unit
POCC	Payload Operation Control Center
PSD	Power Spectral Density

PSS	Payload Specialist Station
RAU	Remote Acquisition Unit
RCS	Reaction Control System
RF	Radio Frequency
RMS	Remote Manipulator System
RPA	Retarding Potential Analyzer
RT	Real Time
SCO	Subcarrier Occillator
SCT	Silver Coated Teflon
SIPS	Small Instrument Pointing System
S/L	Spacelab
SPF	Spacelab Processing Facility
STDN	Space Tracking and Data Network
STEM	Steerable Tubular Extendable Members
STS	Space Transportation System
TCS	Thermal Control System
TDRSS	Tracking and Data Relay Satellite System
TDRST	Tracking and Data Relay Satellite Terminal
TQCM	Thermal Controlled Quartz Crystal Microbalance
TRASYS	Thermal Radiation Analysis System
UV-VIS-IR	Ultraviolet-Visible-Near Infrared
VAB	Vertical Assembly Building
VCO	Voltage Controlled Oscillator
VLF	Very Low Frequency
XPOP	X-Axis Perpindicular to Orbital Plane
ZLV	Z-Axis Local Vertical